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NASA Contractor Report 3352

A System for Aerodynamic Design and Analysis of Supersonic Aircraft

Part 2 - User's Manual

W. D. Middleton, J. L. Lundry,
and R. G. Coleman

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A System for Aerodynamic Design and Analysis of Supersonic Aircraft

Part 2 - User's Manual

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and R. G. Coleman
Boeing Commercial Airplane Company
Seattle, Washington

Prepared for
Langley Research Center
under Contract NAS1-15534

National Aeronautics
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1.0 SUMMARY

An integrated system of computer programs has been developed for the design and analysis of supersonic configurations.

The system consists of an executive driver and eight basic computer programs including a plot module, which are used to build up the force coefficients of a selected configuration. Documentation of the system has been broken into four parts:

Part 1 - General Description and Theoretical Development (CR-3351)
Part 2 - User's Manual (CR-3352)
Part 3 - Computer Program Description (CR-3353)
Part 4 - Test Cases (CR-3354).

This part, the user's manual, contains a description of the system, an explanation of its usage, and the input definition.

These four documents supersede NASA contractor reports CR-2715, CR-2716, and CR-2717, which described an earlier version of the system.

Interactive graphics for use with the system are optional, employing the NASA-LRC CRT display and associated software. A description of the interactive graphics portion of the system is given in the Appendix.

The computer program is written in FORTRAN IV for a NOS operating system and library file. It is designed for the CDC 6000 series of computers and is overlay-structured. The system requires approximately 135000_8 central memory words and uses eight peripheral disk files in addition to the input and output files.

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2.0 INTRODUCTION

A series of individual computer programs for design or analysis of supersonic configurations has been linked together into a single system. The system, because of built-in communication between the programs, is substantially simpler to input and use than the individual programs operating in a stand-alone mode. In addition, a common geometry format, based on the NASA-LRC configuration plotting program, has been adopted to standardize the input requirements of the basic programs.

Interactive graphics have been included in the system, to display or edit input and to permit monitoring and readout of program results. The graphics arrangement is tailored specifically to the NASA-LRC CDC 250 cathode ray tube and associated software. However, all graphics applications have been subroutined to the main programs and could be readily converted to a different graphics setup.

Because the segment of the computer program which calculates skin friction drag (page 71) requires input of altitude in thousands of feet and Reynolds number in millions per foot, SI units have not been used.

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3.0 DISCUSSION

A schematic of the design and analysis system is shown in figure 3.0-1. The system consists of an executive "driver" and eight basic computer programs including a plot program and a geometry input module, which are used to build up the force coefficients of a selected configuration as shown in figure 3.0-2. The system may be used with or without interactive graphics.

The complete design and analysis system is a single overlaid computer program, with the executive driver as the main overlay and the basic programs as primary overlays. The basic programs manipulate input (geometry module), draw a picture of the configuration (plot module), or perform design or analysis calculations.

Aerodynamic force coefficients for a selected configuration are built up through superposition. The individual modules of the system provide data for the force coefficient buildup as follows:

- Skin friction is computed using flat plate turbulent theory.
- Wave drag is calculated from either near-field (surface pressure integration) or far-field (supersonic area rule) methods. The near-field method is used primarily as an analysis tool, where detailed pressure distributions are of interest. The far-field method is used for wave drag coefficient calculations and for fuselage optimization according to area rule concepts.
- Drag-due-to-lift is computed from the lift analysis program, which breaks arbitrary wing/fuselage/canard/nacelles/horizontal tail configurations into a mosaic of "Mach-box" rectilinear elements which are employed in linear theory solutions. A complementary wing design and optimization program, also using the Mach-box approach, solves for the wing shape required to support an optimized pressure distribution at a specified flight condition.

3.1 System Communications

Communication between the executive and the different basic modules is performed by disk files and limited common block storage.

1) Input

   All input to the basic modules is handled through the common geometry module and its associated interfaces. A fundamental consideration in the setup of the system has been that input to the basic modules would not be changed by their incorporation into the overall system. However, to minimize and simplify system input requirements, a special geometry module has been created to read all input, and then sort and structure the input needs of the basic programs.

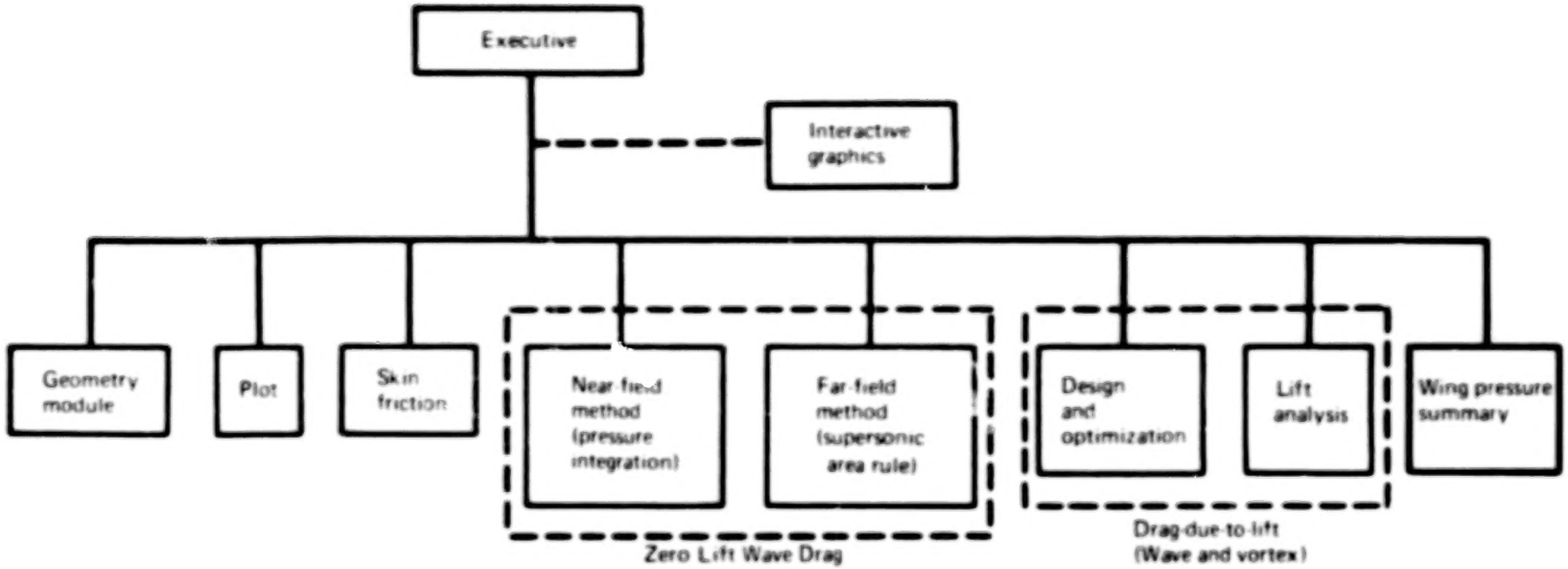
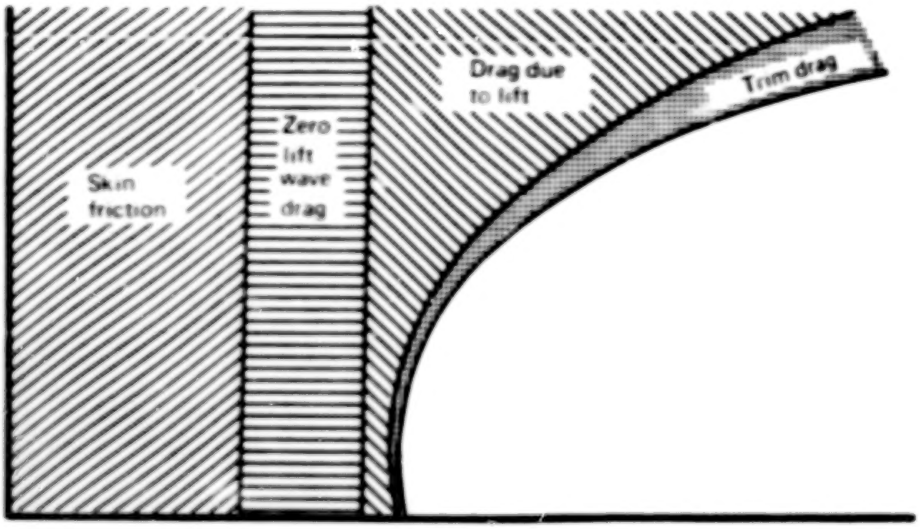


FIGURE 3.0-1.—INTEGRATED SUPERSONIC DESIGN AND ANALYSIS SYSTEM

SUPERPOSITION METHOD OF DRAG ANALYSIS



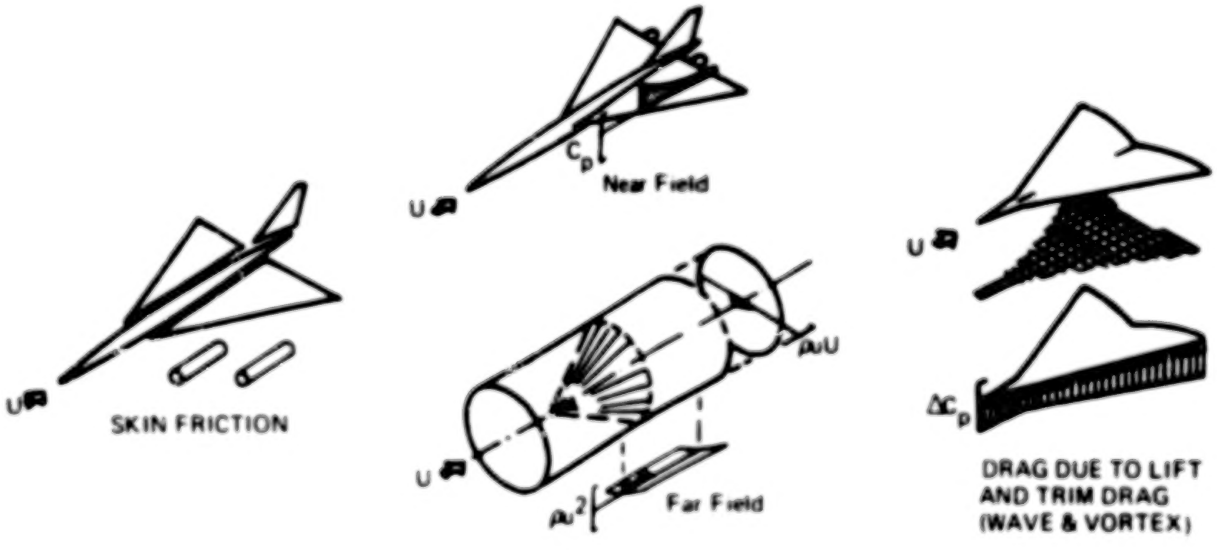


FIGURE 3.0-2.–DRAG BUILDUP

2) Program Sequencing

Program execution is ordered by means of special identification cards, read in the executive, which initiate a specific operation; for instance:

GEOM This card instructs the executive to have the geometry module read configuration geometry.

PLOT This card orders a plot of the configuration to be drawn, according to size and view requirements which will be supplied.

SKFR Compute skin friction for the configuration.

Other similar cards control the other basic modules. The configuration that is to be plotted, or analyzed, need not be the complete configuration that has been input. Also, the geometry definition may be updated without complete replacement of the geometry input.

A summary of the executive control cards is given in Section 4.

For each basic program, there are some inputs that are not geometry (e.g., Mach number, number of longitudinal cuts in analysis, etc.). These inputs are given immediately after the program calling card and are read in the proper interface routine in the geometry module.

3) Program Answers

A limited amount of common storage between the different programs is used to preserve answers and transfer data between modules. The lift analysis module is the largest single program in the system. Therefore, some common blocks used in the lift analysis program are carried also in the executive level without increasing total system size. These data blocks include:

- Wing camber surface definition
- Wing thickness pressures
- Fuselage upwash buoyancy pressures
- Nacelle pressure field
- Asymmetric fuselage buoyancy field (non mid-wing configurations)

Another data block transfers the optimized fuselage area distribution, based on wave drag considerations, to the geometry module for updating.

Wing pressure data for use in the pressure summary module (WPLT) utilize the above common blocks and two supplemental disk files for data storage.

3.2 Geometry Module

The function of the geometry module is to read system geometry input, update it if required, and arrange it as needed for the individual programs of the system. A schematic of the geometry module is shown in figure 3.2-1.

The geometry module is accessed by the executive control cards GEOM NEW (input new configuration) or GEOM (addition or replacement of components). The geometry module is also called to update the fuselage or wing camber surface definitions if the executive cards FSUP or WGUP are read.

In addition, the geometry module is called by the executive as an intermediate step in the execution of any of the basic programs. This requires the proper interface routine to be entered, the system geometry to be put into the correct form for the program to be executed, and any special (non-geometric) data required to be read. This is all stacked in the proper order, whereupon the executive then calls the basic program.

In order to minimize core storage requirements of the input data, both the basic system geometry and the transferred input (from the geometry module to another program) are stored on tape (or disk). The basic system geometry is preserved on a tape when the geometry module is not in core, and the input "stack" for a given program is written on a tape to be read by the programs when called by the executive. The input tape created by the geometry module thus merely replaces the usual input tape written from cards.

The format of the system geometry input is the same as that of the NASA-LRC plot program (reference 2). There are some restrictions (relative to the reference 2 input) in the allowable number of input defining stations, however. The geometry format and limits are given in section 4. Some optional geometry has also been added. This includes provisions for fuselage perimeters to be input (if needed by the skin friction program), and provisions for wing camber surface input at planform spanwise stations other than those specified for the system geometry. This camber surface definition, called WZORD, is data in the form normally generated or used by the wing design and analysis programs. Also, nacelles may be located either in the Z coordinate system of the basic geometry or relative to the local wing surface, whichever is more convenient.

3.3 Plot

The plot module generates the necessary instructions for drawings of the input configuration, either in hard-copy form (Cal Comp) or on the cathode ray tube. Various view options are available. The view option and drawing size are controlled by program inputs.

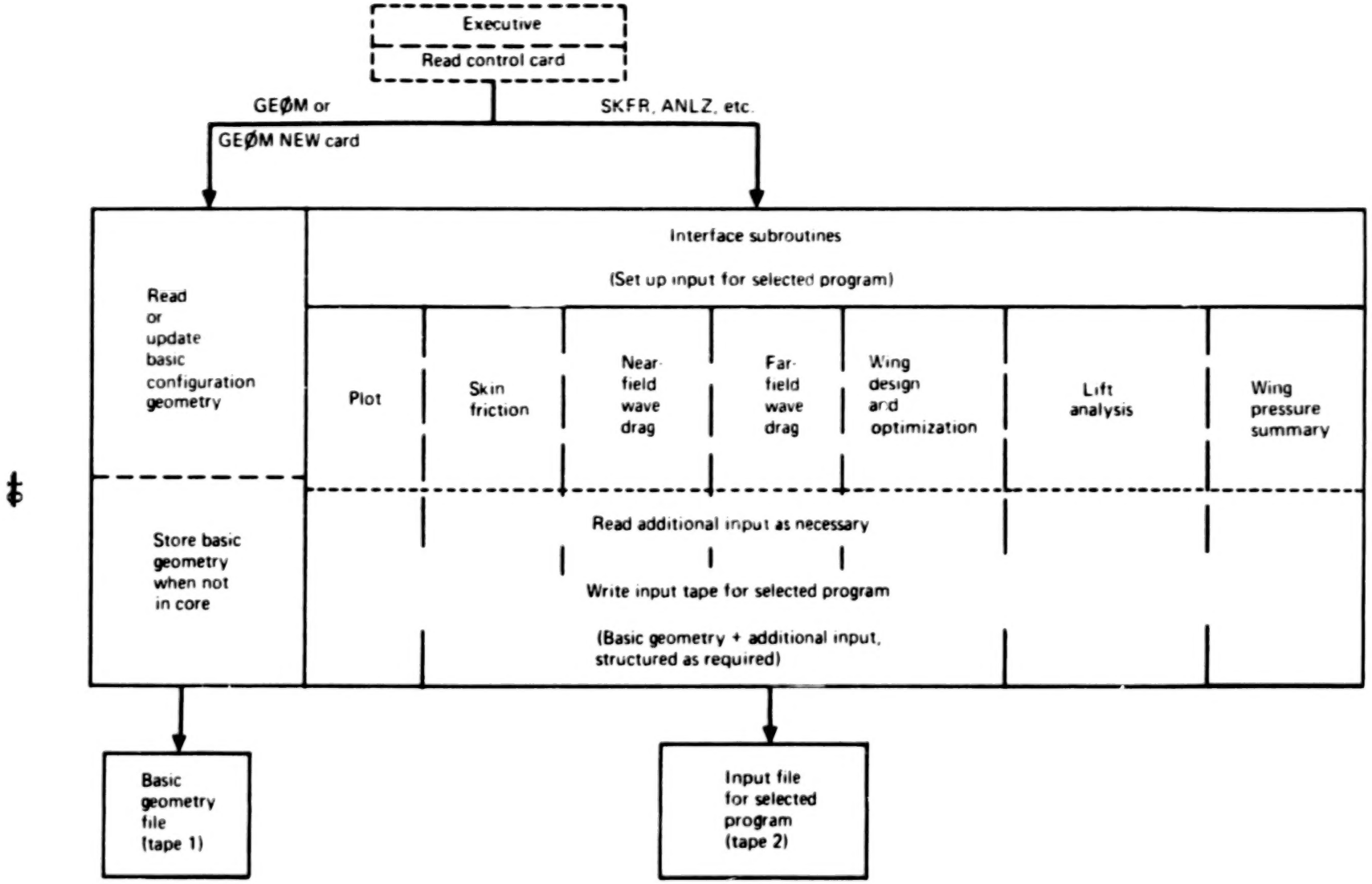


FIGURE 3.2-1.—SCHEMATIC OF GEOMETRY MODULE

The plot program was developed at NASA-LRC and has been incorporated into the system with minimum change. Documentation of the program is presented in reference 2.

A typical configuration drawing generated by the plot program is shown in figure 3.3-1.

3.4 Skin Friction

Skin friction drag for a configuration is computed by separating the airplane into its components, then calculating wetted area and the corresponding turbulent skin friction drag for each component. The wing, tail and/or canard (components which may have large variations in chord length) are strip-integrated to obtain an accurate average skin friction coefficient. Skin friction coefficients are computed from the method of reference 1.

Flight conditions for skin friction calculations may be input either as Mach number/altitude, or Reynolds number per foot and total temperature. If the user wishes to input wetted areas for the different components, rather than have the program generate the wetted areas from the system geometry, several special input options are provided.

A schematic of the skin friction program is shown in figure 3.4-1.

3.5 Far-Field Wave Drag Program

This program computes the zero-lift wave drag of an arbitrary configuration by means of the supersonic area rule. The program was originally developed at the Boeing Company, and has been documented (reference 3) and updated by NASA-LRC. The version of the program used in the design and analysis system is that of LRC.

The far-field wave drag program is extremely versatile, and includes a fuselage area optimization feature which is very useful. The fuselage optimization is accomplished by requiring the program to optimize the overall area distribution of wing-nacelles-tail, etc., subject to a few fuselage area control points or "restraints". The program then fills-in the non-restrained fuselage area distribution in an optimum fashion for minimum wave drag.

In the design and analysis system, a fuselage area distribution may be optimized by initially defining it in the basic geometry, optimizing the definition in the far-field wave drag program, and then transferring the optimized definition to the geometry module for use in further design or analysis cycles. The actual transfer of the optimized fuselage geometry is performed by use of the executive card FSUP, as described in Section 4.

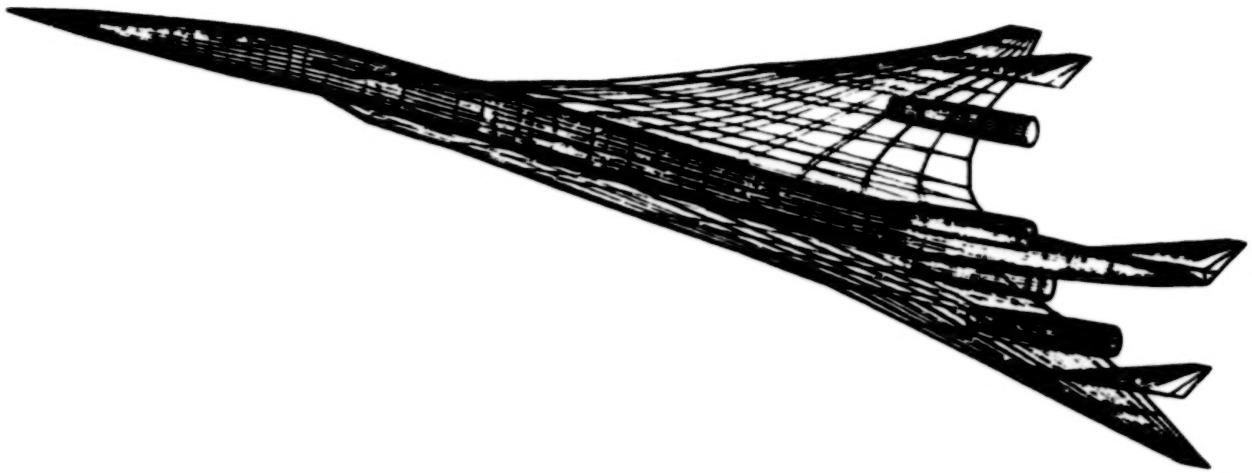


FIGURE 3.3-1.—TYPICAL PLOT PROGRAM DRAWING

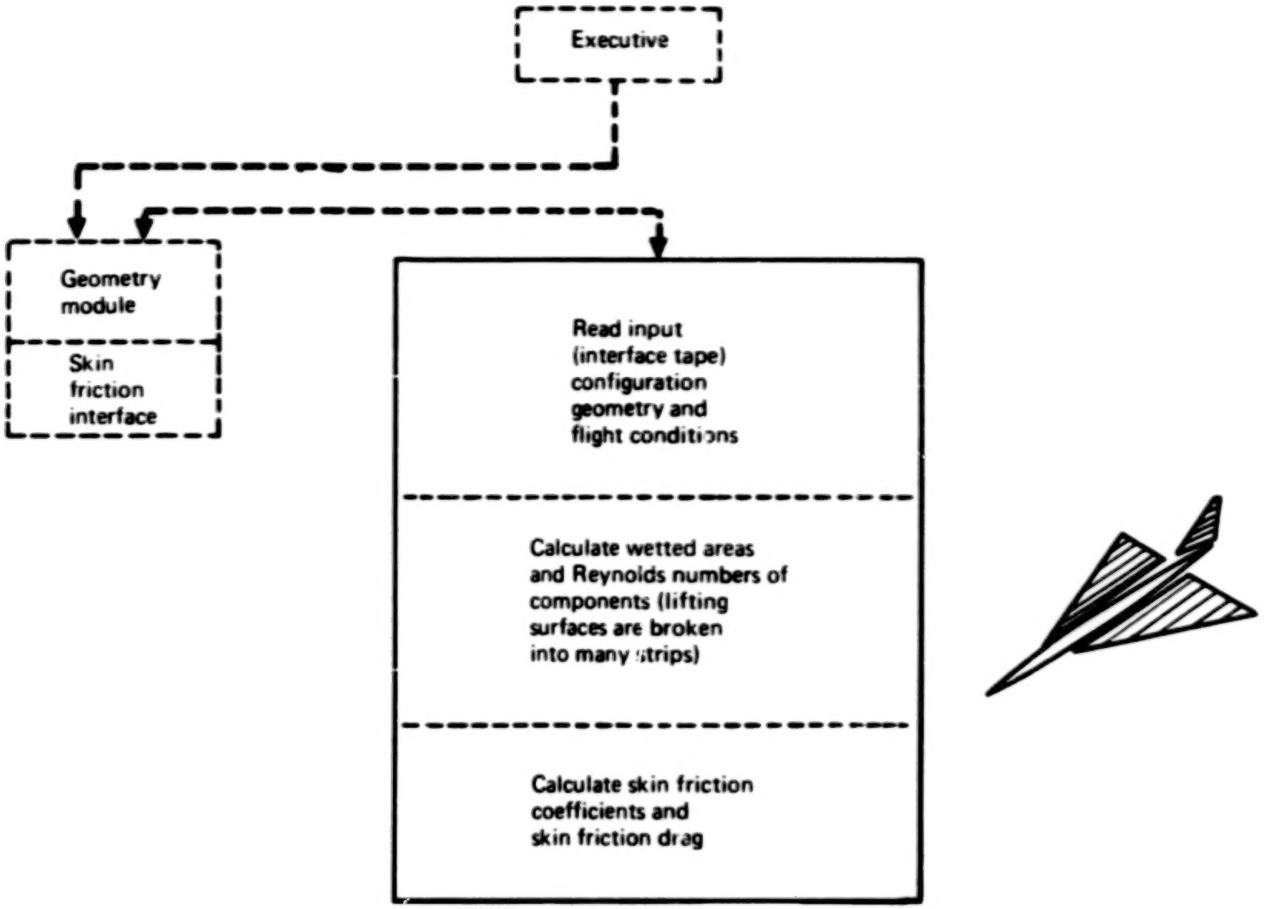

Executive
Geometry module
Skin friction interface
Read input (interface tape) configuration geometry and flight conditions
Calculate wetted areas and Reynolds numbers of components (lifting surfaces are broken into many strips)
Calculate skin friction coefficients and skin friction drag

FIGURE 3.4-1.–SCHEMATIC OF SKIN FRICTION PROGRAM

3.6 Near-Field Wave Drag

The near-field wave drag program computes zero-lift thickness pressure distributions for an arbitrary wing-body-nacelle configuration. The pressure distributions are integrated over the cross-sectional areas of the configuration to obtain the resultant drag force. This force may or may not correspond directly to the drag computed by the far-field method, depending upon the degree of transparency specified for the near-field pressure integrations.

By transparency is meant the assumption of the far-field method that pressure fields from all components "pass through" and interact with all other components, regardless of possible physical barriers imposed by in-between components.

Typical pressure data from the near-field program are presented in figure 3.6-1. A wave drag coefficient summary from the program is shown in figure 3.6-2.

The near-field program has three principal uses:

1) As an analysis tool for studying the zero-lift drag forces associated with the interacting pressure fields of different configuration components. In this respect, the near-field program has an advantage over the far-field wave drag method in that there need be no assumption of transparency.

2) As a source of loads data for structural design and analysis.

3) As a source of thickness pressure fields for use in the pressure limiting options of the wing design and lift analysis programs. (This option is described in section 3.7, but basically requires that the total surface pressure coefficient on the wing, i.e., thickness plus lift, cannot be less than some specified fraction of vacuum pressure coefficient.)

   If the wing thickness pressures are to be used by the wing design or lift analysis programs in pressure limiting options, then the near-field program must first be run. During program execution, the thickness pressures are loaded into a system common block and are then available where needed.

Nacelle pressure field options. - The near-field program allows for up to 3 pairs of nacelles located external to the wing-fuselage (or 2 pairs plus a single nacelle at Y=0). The nacelles may be either above or below the wing (or both).

The nacelle pressure field is the pressure field imposed on the surface of the wing by the nacelles. A feature of the near-field program is the choice of

$C_{D_{wing}} = 0.00113$ $C_{D_{wing\text{-}on\text{-}body}} = 0.00008$ $\Sigma C_{D_{wing\text{-}body}} = 0.00153$

$C_{D_{body}} = 0.00032$ $C_{D_{body\text{-}on\text{-}wing}} = -.000003$

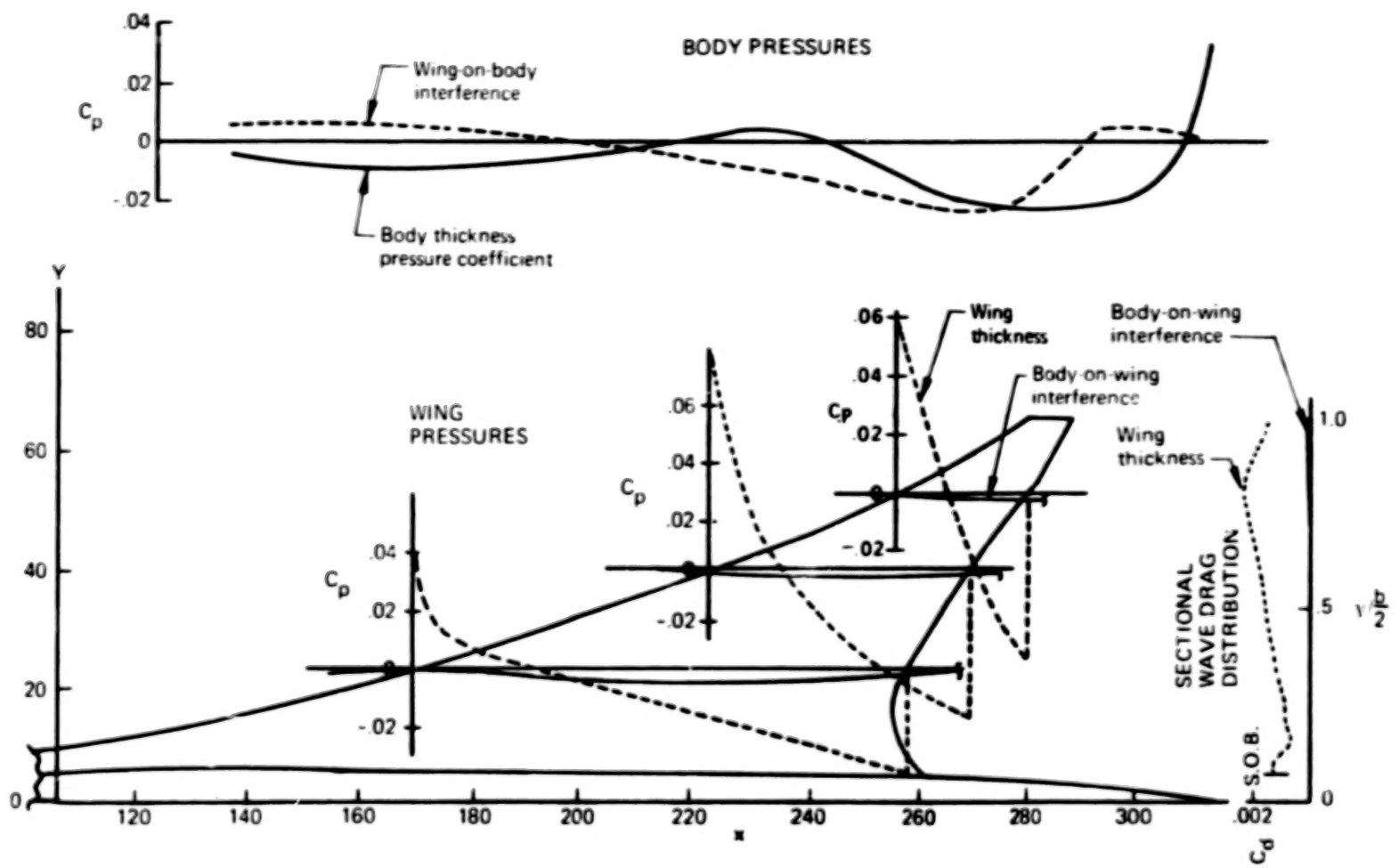
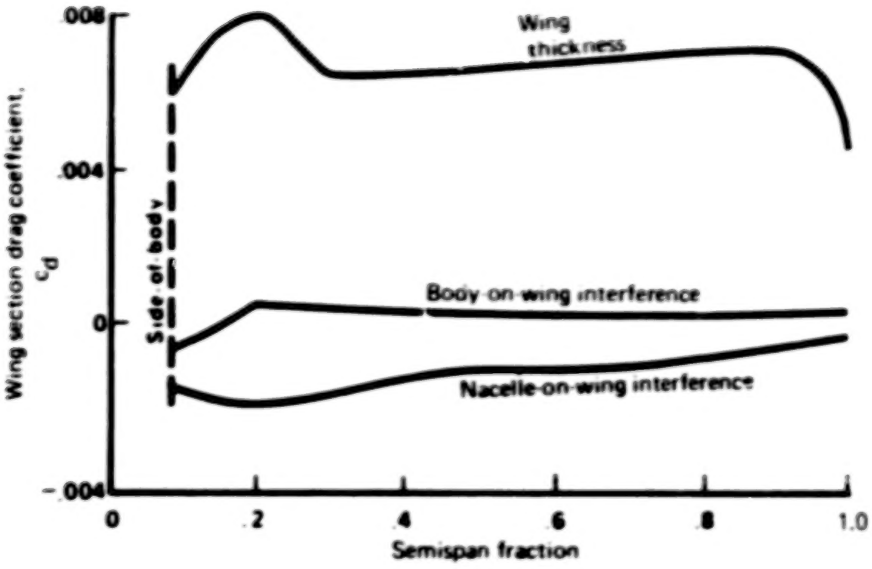


FIGURE 3.6-1.–WING-BODY SOLUTION, M = 2.6



Wing-Body Terms

$C_{D_{wing}} = 0.00639$ $C_{D_{wing\text{-}on\text{-}body\ interference}} = -0.00013$

$\Sigma = 0.00711$

$C_{D_{body}} = 0.00072$ $C_{D_{body\text{-}on\text{-}wing\ interference}} = 0.00013$

Nacelle Terms

	Inboard	Outboard
Isolated $C_{D_{wave}}$	0.00075	0.00075
Body-on-nacelle interference	-0.00002	0.00000
Nacelle-on-body interference	0.00005	0.00010
Nacelle-on-nacelle interference		
Direct	0.00034	0.00023
Image	0.00054	0.00046
Wing-on-nacelle interference	-0.00043	-0.00058
Nacelle-on-wing interference	-0.00156	

$\Sigma C_{D_{nac}} = 0.00064$

Σ Wing-body-nacelle $C_{D_{wave}} = 0.00775$

FIGURE 3.6-2.–TYPICAL WAVE DRAG COEFFICIENT SUMMARY–NEAR-FIELD PROGRAM (M = 1.1)

"wrap" or "glance" solutions for the nacelle pressure field, as shown in figure 3.6-3. (The far-field wave drag program uses essentially the "wrap" solution).

Available experimental data do not make it clear whether a "wrap" or "glance" solution is more correct. Since the nacelle-on-wing interference term is substantial, both solutions are available in the program (controlled by an input code).

3.7 Wing Design and Lift Analysis

The wing design and lift analysis programs are separate lifting surface methods which solve the direct or inverse problem of:

- Design - to define the wing camber surface shape required to produce a selected lifting pressure distribution. The wing design program includes methods for defining an optimum pressure distribution.

- Lift analysis - to define the lifting pressure distribution acting on a given wing camber surface shape, and calculate the associated force coefficients.

The lift analysis program contains solutions for the effect of fuselage, nacelles, canard and/or horizontal tail, and wing trailing edge flaps or incremental wing twist. Using superposition, the program solves for drag-due-to-lift, lift-curve slope, and pitching moment characteristics of a given configuration through a range of angles of attack at a selected Mach number.

The wing design program is more limited in scope, since it is used to solve for the wing shape required to support a design pressure distribution at a specified flight condition. The program also contains, however, a number of optional features for identifying the design pressure distribution. This is a demanding solution, because it requires that:

- Drag-due-to-lift of the wing be minimized at a given total lift, subject to an optional pitching moment constraint.

- Constraints be applied to the design pressure distribution to provide physical realism.

- Effects of fuselage upwash, nacelle pressure field, etc., be reflected in the design solution.

As a special case, the optimization feature of the wing design program may be bypassed and the wing designed to support a single input loading definition. This may be done either with or without including the effects of fuselage upwash, nacelle pressure field, etc.

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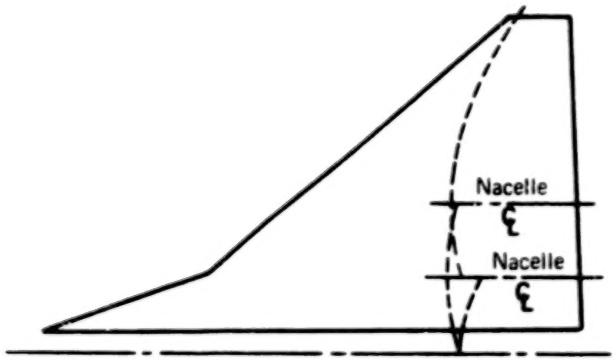
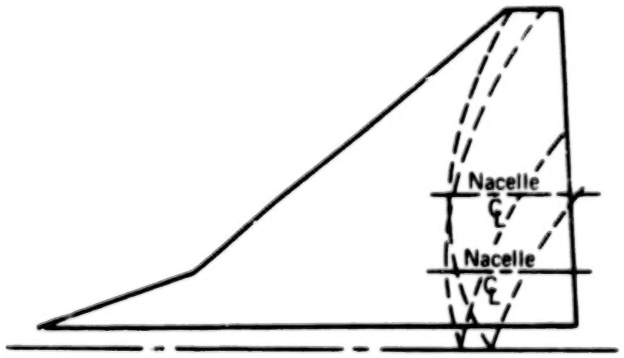


PRESSURES "GLANCE" AWAY FROM WING AT ADJACENT NACELLES

The nacelle pressure field and accompanying shock waves "glance" away from the wing when encountering adjacent nacelles. In application, the nacelle generated pressure field is terminated on encountering another nacelle.



PRESSURES "WRAP" AROUND ADJACENT NACELLE

The nacelle pressure fields and accompanying shock waves "wrap" around adjacent nacelles. In application, the nacelle generated pressure field is allowed to pass through another nacelle as if it were transparent.

FIGURE 3.6-3.—NACELLE PRESSURE FIELD CONCEPTS

Wing Design and Optimization

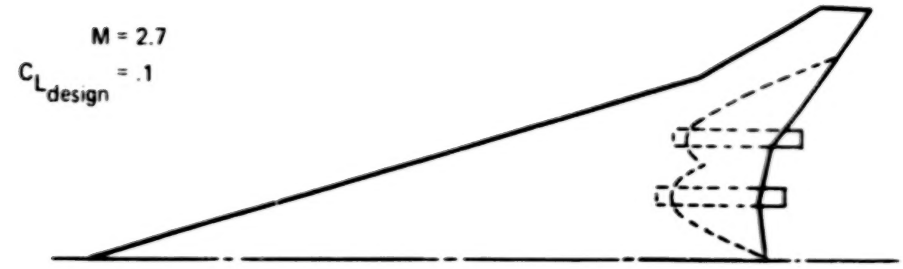
Given a wing planform and flight condition, the wing design program solves for an optimum (least drag) pressure distribution and the corresponding wing shape, subject to specified constraints on:

- Total lift
- Pitching moment at zero lift
- Upper surface pressure coefficient level and/or streamwise gradient
- Ordinate at defined planform locations

Basically, the method of the wing design program is that of references 4 and 5. For use in the integrated design and analysis system, however, the program has been expanded to provide the following capability:

- Use of any combination (or all) of ten basic lifting pressure loadings, in an optimum fashion.
- Optional imposition of pressure level and pressure gradient constraints on the wing upper surface, to prevent occurrence of unrealistically low pressure coefficients.
- Optional consideration of three configuration-dependent loadings (fuselage upwash and buoyancy, and nacelle pressure field).
- Optional consideration of three wing camber-induced loadings which are proportional to the three configuration-dependent loadings, This introduces camber-related terms to modulate the configuration related loadings (Example: trailing edge reflex for nacelle buoyancy loading).
- Optional identification of a small planform region (e.g., trailing edge flap) for special incremental loading.
- Optional constraints on camber surface ordinate at specified planform locations.

The presentation of the wing design results, for selection of an optimum pressure distribution, is in the form of drag-due-to-lift versus zero-lift pitching moment (C_{mo}). A typical presentation is shown in figure 3.7-1, illustrating the effect of increasing the number of design loadings and adding the nacelle-buoyancy loading. Selecting a C_L and C_{mo} combination for the wing defines a corresponding pressure distribution which may then be used to generate the associated wing camber surface shape. (The bucket plot is not used with ordinate constraints, however, and only the solution corresponding to the design point values of C_L and C_{mo} is printed.)



Note:

At the design points denoted by circular symbols,

$$C_{p_{\substack{upper \\ surface}}} \geq 0.7\, C_{p_{vacuum}}$$

Wing thickness pressures included

Two and three loading combinations are the first two and first three loadings in Table 1

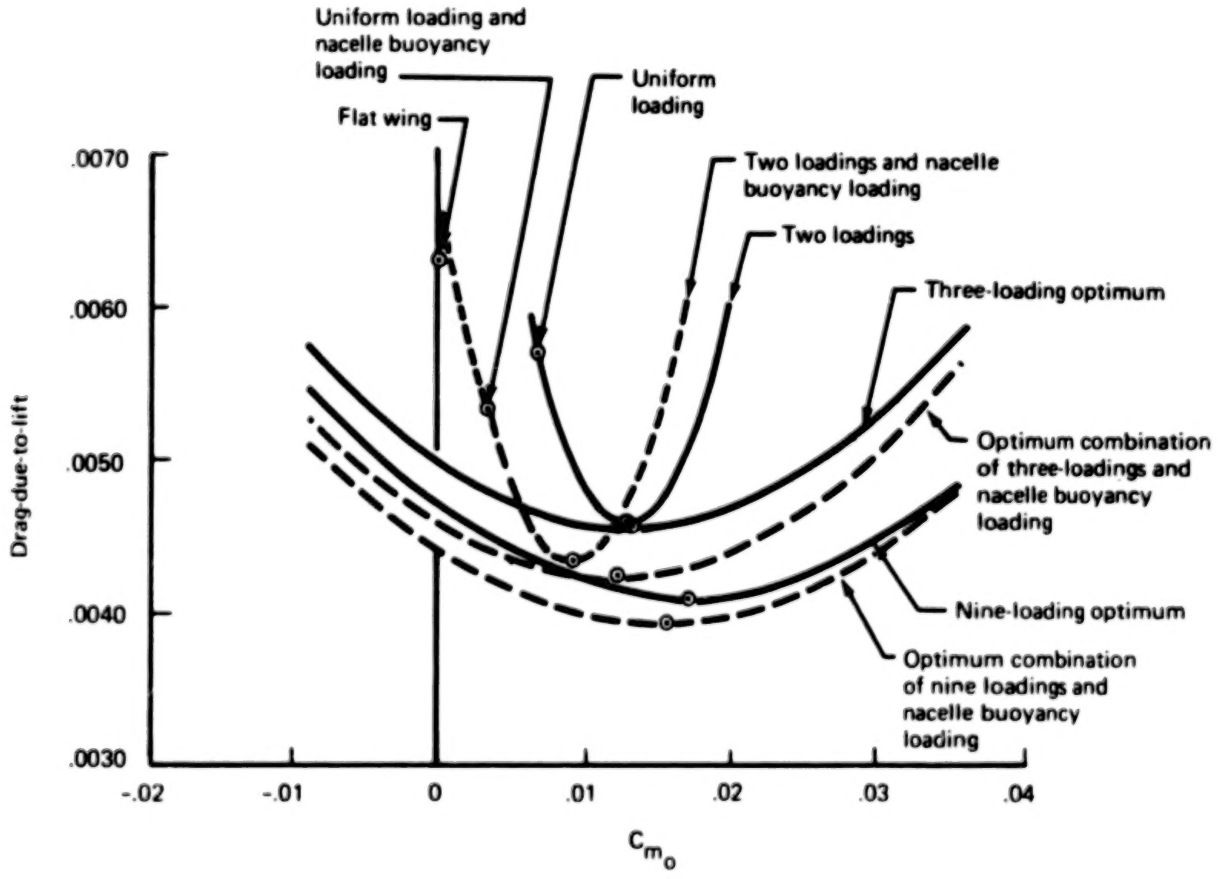


FIGURE 3.7-1.–EFFECT OF NUMBER OF LOADINGS ON WING DESIGN

Pressure constraints. - The use of a large number of basic wing loadings permits great flexibility in identifying a theoretically optimum lifting pressure distribution. Such an optimum may be physically unrealistic, however. Linear theory contains no limitations on allowable surface pressures, i.e., "optimum" pressure distributions may well involve upper surface pressure coefficients lower than vacuum C_p. To avoid this possibility, a pressure constraint formulation has been added to the solution. This functions by limiting the total wing upper surface pressure coefficient to be equal to or greater than an input C_p, and by limiting the longitudinal gradient of this upper surface pressure to be less than or equal to an input gradient level.

By superposition, the total upper surface pressure coefficient is the sum of wing thickness pressure (from the near-field wave drag program, as noted in Section 3.6), fuselage pressure field, and the upper surface lifting pressure.

The effect of constraining the allowable design pressure distribution for a basic wing planform (no fuselage) is illustrated in figure 3.7-2. For a given planform and set of loadings, the program cycles to find an optimum pressure distribution (least drag) subject to input constraint conditions. First an optimum loading combination is found, then the corresponding peak pressure level and gradient are located. If either violates the input limits, a new optimum loading is found with a pressure constraint applied at the location of the maximum pressure violation. The new optimum is then examined, etc. Gradient is everywhere satisfied before level is constrained, as described in the theory document, volume 1.

The cyclic operation continues until the wing pressure distribution everywhere satisfies the pressure constraints. In the example case shown, the effect of adding pressure constraints shifts the drag minimum from the bucket plot level to the level indicated by the flagged symbol.

It can occur that the input pressure gradient constraint cannot be satisfied within the other constraint bounds of C_{mo}, wing thickness pressures and/or Z constraints. In this case, the program automatically increases the input acceptable gradient level by 20 percent and tries again. This process will continue until the gradient level is satisfied. No similar option is applied to the pressure level constraint, however; if pressure level cannot be satisfied, the program halts.

A further discussion of pressure constraint application is given on page 32.

Loading definitions. - A tabulation of the pressure loadings available within the design program is given in Table 1 on page 23. The configuration dependent loadings may be used both as a superimposed, independent effect and also as a definition of a loading which may be varied (by wing camber) in the optimization process.

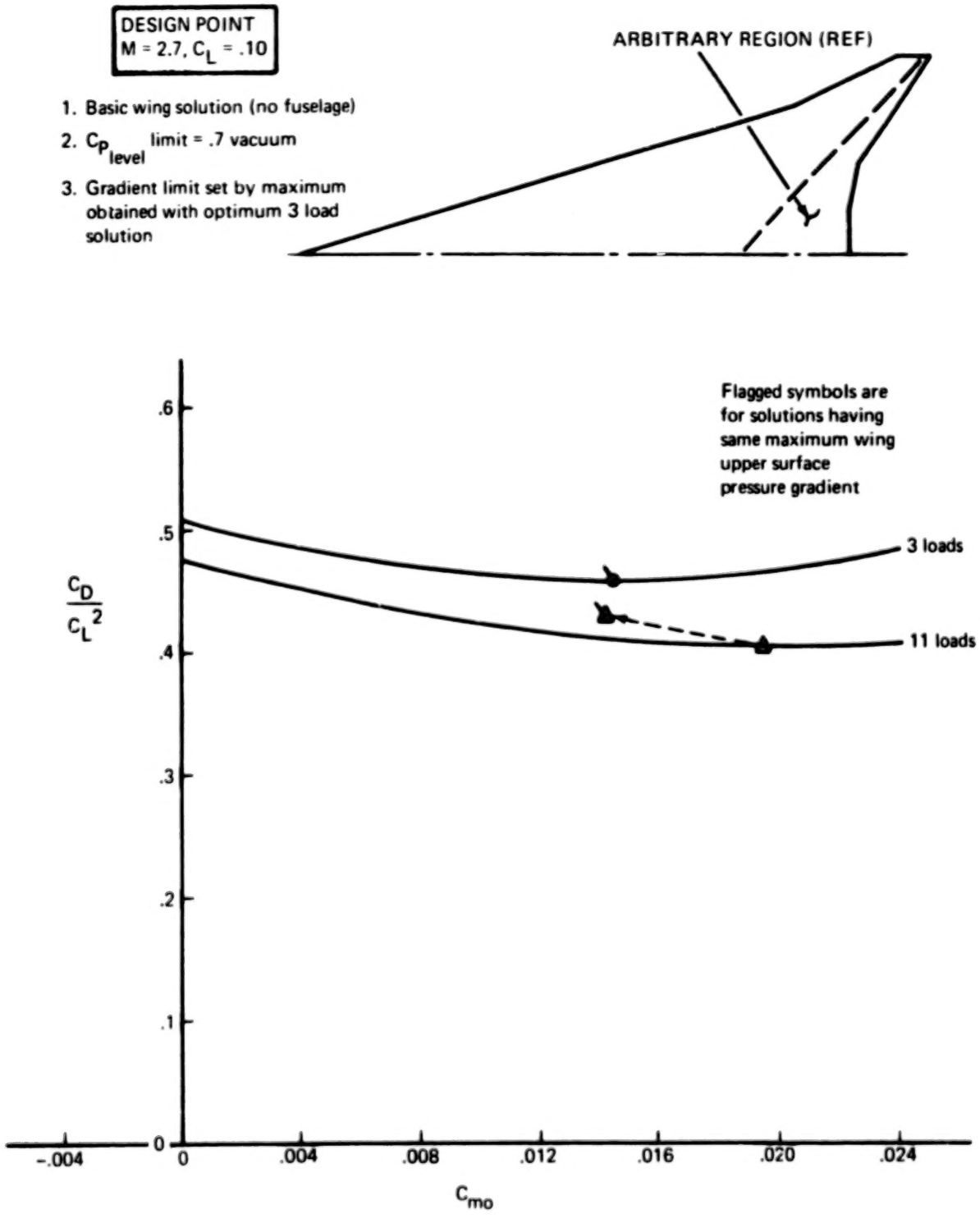


FIGURE 3.7-2 – EFFECT OF PRESSURE CONSTRAINTS

TABLE 1.–DESCRIPTION OF WING LOADING TERMS

Loading Number	Definition
1.	Uniform
2.	Proportional to x, the distance from the leading edge
3.	Proportional to y, the distance from the wing centerline
4.	Proportional to y^2
5.	Proportional to x^2
6.	Proportional to x(c - x), where c is local chord
7.	Proportional to x^2 (1.5 c -x)
8.	Proportional to $2\,(1 + 5\,\frac{x}{c})^{-0.5}$
9.	Proportional to $\eta^2\,(\eta - 1)^2$
10.	Elliptical spanwise, proportional to $\sqrt{(1 - \eta)}$
11.	Proportional to x, the distance from the leading edge of an arbitrarily defined region
12.	A camber-induced loading proportional to the body buoyancy loading
13.	A camber-induced loading proportional to the body upwash loading
14.	A camber-induced loading proportional to the nacelle buoyancy loading
15.	The body buoyancy loading
16.	The body upwash loading
17.	The nacelle buoyancy loading

- As an independent effect, the configuration-dependent loading acts upon the wing in the optimization process, but cannot be varied (loadings 15-17).

- As a loading definition (12-14), a configuration-dependent loading produced by wing camber may be introduced in addition to its independent effect. The optimization then could cancel the lift of the independent effect with this camber-generated loading, if that were the optimum solution.

In general, a configuration-dependent loading may not be used as the source of a variable loading without also using it as an independent loading. An exception is the nacelle pressure field loading, which may be used as a camber surface loading independent of the configuration dependent loading. The reason for this treatment is to allow input of a special loading as the nacelle pressure field, for use either in the optimization process or as a single loading. This feature is discussed on page 30.

Fuselage in wing design solution. - The fuselage may be included in the wing design solution by input of fuselage geometry and specifying a side-of-fuselage semispan station. The resulting solution is then split into two parts: the wing part (outboard of side-of-fuselage station) with loading definitions as described previously, and the "fuselage" part (inboard of side-of-fuselage station). Loadings inboard of the side-of-fuselage station are of the "carry-over" type, and are dependent functions of the loadings outboard of the side-of-fuselage.

Drag of the "fuselage" part is calculated by applying the carry-over loadings to the fuselage camberline. The outboard, or wing, part is handled as for the wing alone case, with integrations beginning at the side-of-fuselage. The wing-fuselage solution thus reflects the interdependence of wing and fuselage contributions to the wing design optimization.

There are several considerations of importance in the wing/fuselage solution:

- Wing paneling (internal definition of wing geometry) may require a slight shift in the input side-of-fuselage station. This is accomplished automatically in the program and an explanatory note is printed.

- The fuselage attitude and wing camberline at the side-of-fuselage must approximately align for the drag integrations to be valid. Experience has shown that it is necessary for Z constraints to be applied to the wing camberline at the side-of-fuselage for this to occur. (Z constraints are discussed in more detail on page 36.)

- For convenience, the fuselage attitude (relative to the basic geometry definition) in both the lift analysis program (to generate

upwash loading) and the wing design program can be changed without revising the basic geometry. (In the lift analysis module, this is done by a special application of the pressure limiting option. Set FLIMIT=1.0, an appropriate value of vacuum fraction VACFR, and define the fuselage angle of attack in TLALP).

Inclusion of the fuselage in the wing design solution is illustrated in figure 3.7-3. In the example case shown, the wing camberline at the side-of-fuselage was constrained at the four locations indicated. A bucket plot is not produced when Z constraints are used; however, the effect of adding pressure constraints to the solution is illustrated by the symbols on the drag-due-to-lift versus C_{mo} plot.

Use of configuration-dependent loadings. - An example of the inclusion of a configuration-dependent loading is illustrated in figure 3.7-4, showing "reflexing" of the wing due to nacelle influences. The wing trailing edge is bent upward locally, or reflexed, to take advantage of positive pressure coefficients from the nacelle pressure field.

The loadings due to the fuselage include both lift caused by the fuselage upwash field and also lift due to asymmetric distribution of fuselage volume above and below the wing (non mid-wing arrangement).

As a special case, the asymmetric fuselage buoyancy loading (number 15), can be used even if its net lift is zero. This feature permits the inclusion of fuselage thickness pressures in the pressure limiting case for a mid-wing arrangement. However, if the fuselage buoyancy lift is zero, the wing camber loading proportional to the fuselage buoyancy loading (number 12) cannot be used, since it would cause the optimization solution to fail.

An option in the wing design program permits card input of the configuration dependent loadings. As calculated by the near-field and lift analysis programs, these loadings are written on a file in the wing pressure summary program (see page 54). This option is provided to allow editing of the configuration dependent loadings and/or input of special loadings in their place.

Optimization of the wing design considering influence of the fuselage upwash field is performed iteratively, using both the wing design and lift analysis modules. A fuselage shape and incidence is first assumed, the upwash field and corresponding loading is calculated by the analysis program, and the design solution is performed. Because the resulting wing shape probably differs from the shape used in the initial upwash solution, the upwash loading used is approximate. It may be desirable to then rerun the wing design solution and/or alter the fuselage angle of attack. A representative program executive card sequence would be:

Event	Executive Card
Define fuselage	GEOM
Calculate upwash loading	ANLZ (WHUP=1.0)
Wing design solution	WDEZ
Recalculate upwash loading with new camber surface	ANLZ (WHUP=1.0, TIFZC=3.0)
Wing design	WDEZ

When the wing camber surface is finalized, it may be transferred into the basic geometry by the executive control card WGUP. (With interactive graphics attached, the design wing shape may also be viewed and edited between design and analysis solutions.)

Nacelle pressure field options. - Special options are available for use with the nacelle pressure field (NPF) loading. The NPF is given special treatment because of the character of the NPF and because it is the most general loading that can be input in table lookup format. (This general input format makes the NPF useful as a means of inputting a desired loading which may not be the nacelle pressure field.)

The typical nacelle pressure field, as calculated by the near-field or analysis programs, is characterized by pressure discontinuities associated with shock waves in the flow (e.g., figure 3.7-4). When the NPF is then used to calculate the corresponding camber surface, undesirable discontinuities are required in the surface shape. An option is provided to alleviate this difficulty, which preserves the general nature of the NPF but substitutes an approximate (smoothed) NPF for the camber surface calculation. This option is controlled by the input code FNPSMO. A typical improvement in camber surface smoothness is shown in figure 3.7-5. Regardless of the code FNPSMO, the configuration-dependent loading of NPF (loading 17), if used, uses the NPF without smoothing.

The nacelle pressure field, like the other configuration-dependent pressure tables, may be input into the wing design program to define a special loading to be used either alone or with other loadings. In this usage, the configuration-dependent loading version of the NPF (loading 17) cannot also be used since there is only one definition of the NPF (except for the smooth form noted above, which would not be of interest). In this case, the configuration-dependent version of the NPF could be combined with one of the other configuration-dependent loadings and input in card form (see page 80).

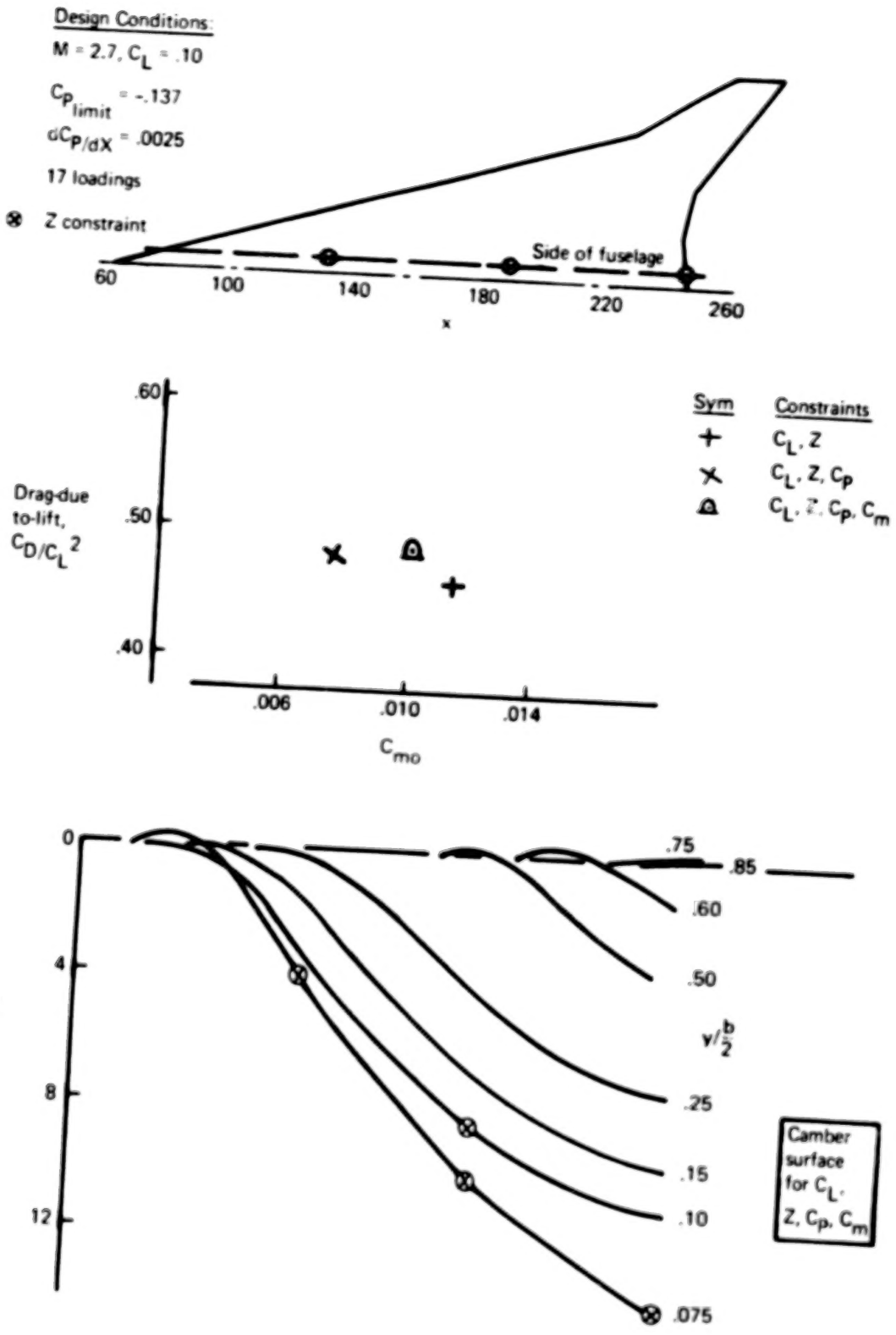


FIGURE 3.7-3.—WING DESIGN OPTIMIZATION WITH FUSELAGE AND Z CONSTRAINTS

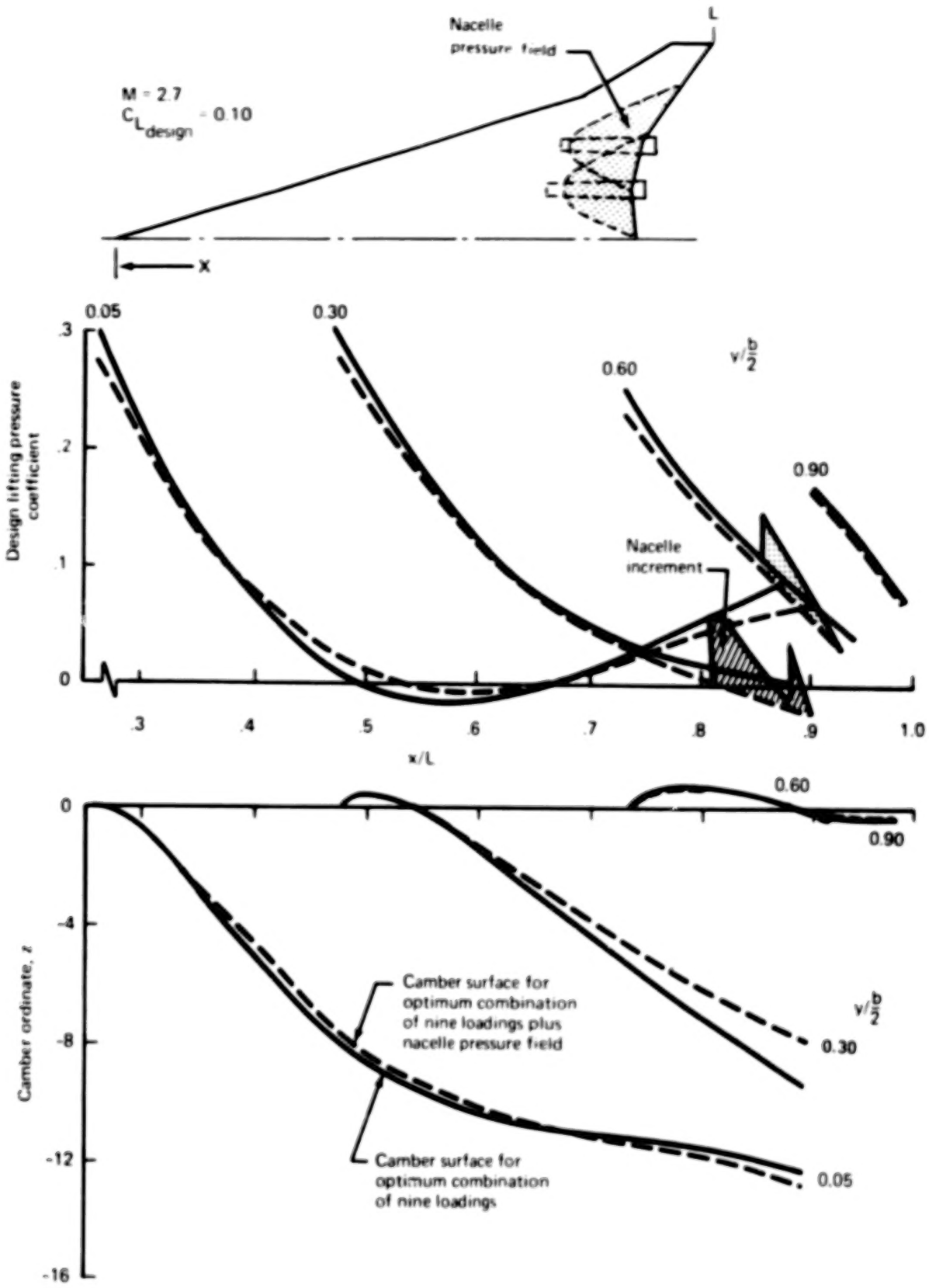


FIGURE 3.7-4.–EFFECT OF ADDING NACELLES TO WING DESIGN SOLUTION

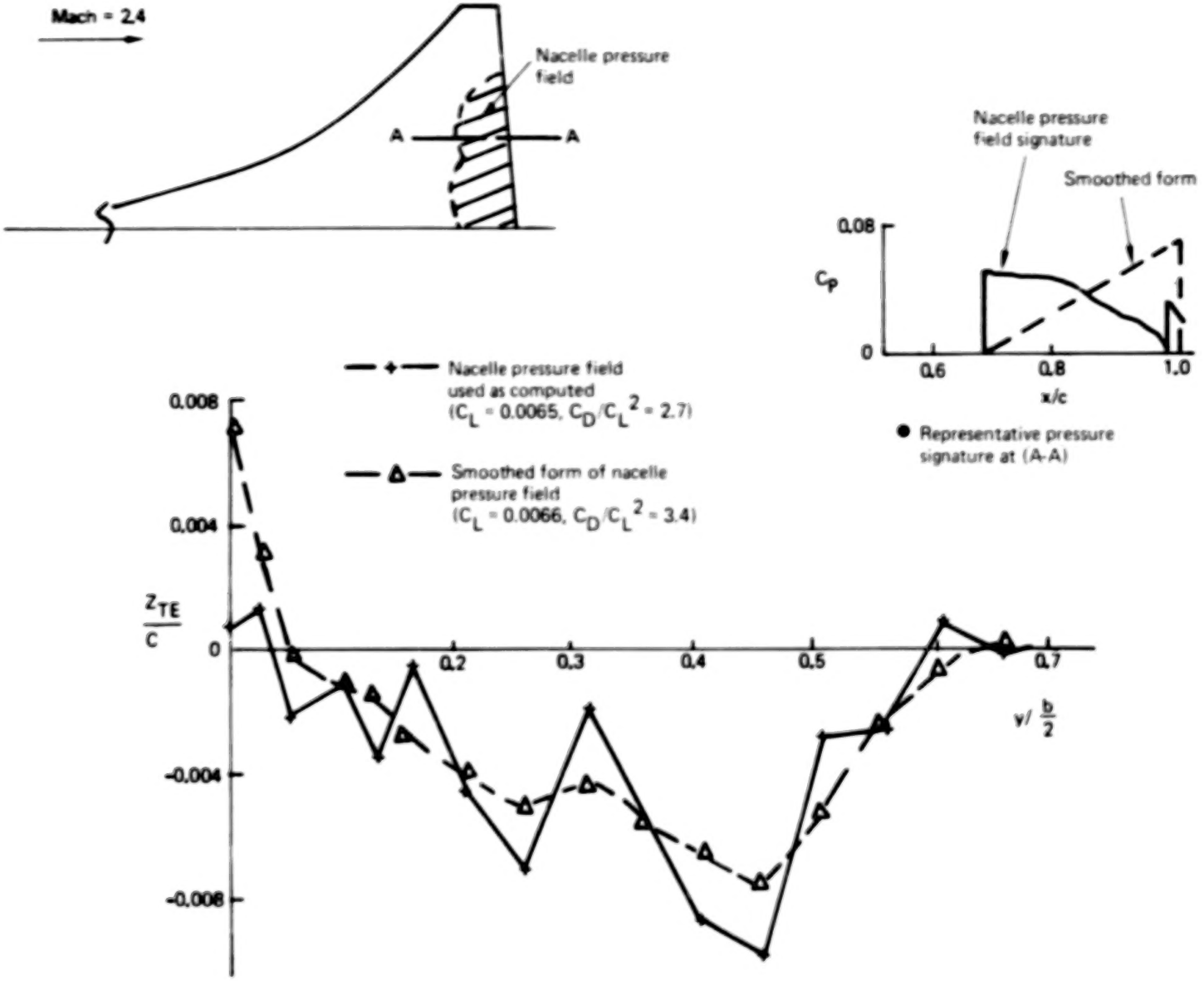


FIGURE 3.7-5.–EFFECT OF SMOOTH FORM NACELLE PRESSURE FIELD ON ASSOCIATED CAMBER SURFACE SHAPE

Single loading option. - The nacelle pressure field input or any of the 10 basic loadings in Table 1 may be used as a single loading, bypassing the optimization feature of the wing design program. This option is requested by the input code SLFLAG. Configuration-dependent loadings (loadings 15-17) that do not produce a camber surface may also be used in combination with the single loading. If the single loading option is selected, no pressure, moment, or Z constraint may be applied. Only lift coefficient may be specified. If C_L = 0. is input, the camber surface will be calculated for the loading as input. Otherwise, the total C_L will be constrained to the input C_L value.

Small planform region option. - Since there may be small regions of the wing (such as a trailing edge flap) that could be relatively highly loaded to good advantage, a program option allows the definition of such a region and a corresponding loading (no. 11 in Table 1).

An example of the use of the planform region option is shown in figure 3.7-6. Inclusion of the region and loading 11 results in a small improvement in drag-due-to-lift, especially as C_{mo} is increased.

A condition imposed upon the planform region option is that the region cannot be re-entrant in the spanwise direction, relative to the forward end. The region is input starting at the most inboard span station (which will be at the wing trailing edge), and successive span stations must increase monotonically.

Loading 11 and the small planform region are only used in combination with each other.

Input considerations. - The wing design program principally requires the specification of a set of loadings, a design point, and the definition of four basic control parameters. The control parameters (on card 7 of the design program input) govern the type and extent of the solution.

The design point solution may be obtained with constraints on:

- C_L only
- C_L and C_{mo}
- C_L and upper surface pressure
- C_L, C_{mo}, and upper surface pressure

If ordinate constraints are specified, they are included with each of these four types of solutions.

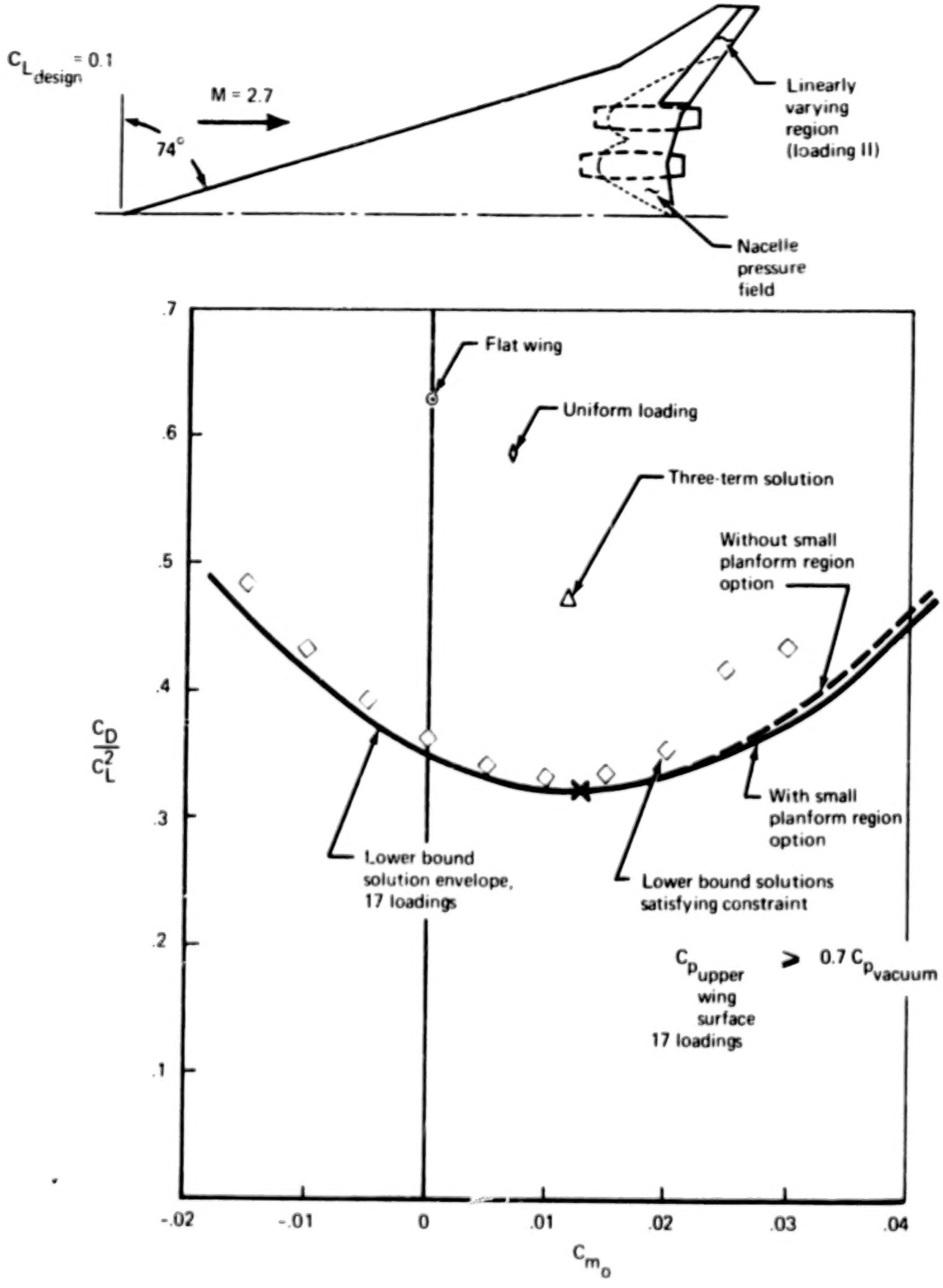


FIGURE 3.7-6.—PLANFORM REGION OPTION

The four types of solutions are not completely independent. If the C_L and constrained pressure solution is requested, then the program must first generate the C_L only solution. Similarly, if the C_L, C_{mo}, and constrained pressure solution is requested, then the program must first generate the C_L and C_{mo} solution. Thus, if the upper surface pressure constraint condition is requested, the program performs the corresponding no pressure constraint solution whether it was requested or not.

It is not necessary to calculate the camber surface shape corresponding to a specific design point (lift coefficient, pitching moment coefficient, constraint condition) in order to obtain the drag-due-to-lift versus C_{mo} plot. Also, if the design camber surface is requested, it may be only printed out, or may be also punched into cards (for later input into the lift analysis program).

Pressure constraint application. - The constraints applied on the pressure distribution of the wing upper surface are of two types. Both pressure level and the longitudinal gradient of pressure can be constrained. This has been done because linear aerodynamic theory can produce pressure distributions (in terms of pressure coefficient) requiring pressure lower than vacuum on the wing upper surface; the first type of constraint can be used to restrict pressure levels to more realistic values. High, positive values of pressure gradient tend to produce boundary layer separation, and the second type of constraint offers a means of control. Constraints on pressure gradient are also quite valuable in controlling wide variations of pressure level.

Pressure constraints of both types are specified by the user with tables of acceptable values (with an optional pressure level constraint definition as discussed on page 34). The tabular format has been chosen so that the values can be varied depending on planform location. After a design solution has been obtained in terms of the component loading factors A_i, the pressure level and gradient on the wing upper surface are compared with the acceptable values. If either or both types of conditions are violated, the program retains the planform locations of the violations in order to apply constraints.

When pressure level violations are detected, the program computes the level of lifting pressure coefficient at the critical planform location, which, in combination with thickness pressure, will produce a wing upper surface pressure equal to the acceptable value. A constrained lifting pressure coefficient set at 95% of the allowable value is then imposed on subsequent solutions for the optimum combination of loadings.

When pressure gradient violations are detected, an acceptable value of pressure gradient due to lifting pressure coefficients is calculated at the critical planform location. Account is taken of the gradient due to wing thickness pressures and any configuration-dependent loadings. A constrained gradient set at 75% of the allowable value is then imposed on subsequent optimizations.

The two factors (95% and 75%) have been applied to diminish a tendency of the solution to vary slightly above the limiting value in the immediate vicinity of the constraint. This type of behavior is most noticeable for cases involving the configuration-dependent loadings, which are specified by tables as a function of planform location and are not necessarily smoothly varying functions.

The application of constraints is cyclic. If the solution without pressure constraints does not satisfy the pressure criteria, a pressure constraint is imposed and a second solution is obtained and tested. If the second solution does not satisfy the criteria, a second pressure constraint is imposed, and a third solution generated and tested, and so forth.

Constraints imposed early in this cyclic fashion can become unnecessary as later constraints are imposed. The solution process includes logic to search for unnecessary constraints after the second and subsequent cycles (the first constraint would not be applied if it were not necessary). At each solution cycle, any unnecessary constraints are removed, and the next cycle has fewer pressure constraints. The constraint identified on the current cycle as most critical is also deleted from the next solution cycle; with fewer pressure constraints, it might no longer be the most critical pressure location.

In summary, the results of each solution cycle are tested in three ways. First, the Lagrange multipliers of the solution corresponding to the pressure constraints are tested. If any constraints are unnecessary, they are removed, and another solution cycle is begun. If all of the pressure constraints are necessary, then the most critical pressure gradient is tested. If it exceeds the allowable value, another gradient constraint is imposed, and another solution cycle is begun. If the solution pressure gradient is satisfactory, then the most critical pressure level is tested. If it exceeds the allowable value, another level constraint is imposed, and another solution cycle is begun. If the most critical pressure level is acceptable, the solution as a whole is acceptable, Solution cycling continues until either an acceptable solution is obtained (cycling stops), or until a limit on the number of constraints is reached.

There are two types of limits on the maximum number of pressure constraints that can be imposed. One limit is imposed by the number of loadings used. The total number of constraints, including those on lift, pitching moment, configuration-dependent loadings, ordinates, and pressure, can at most be equal to the number of loadings. This situation is undesirable for it leaves no degrees of freedom for drag minimization; consequently, a program limit has been set so that two degrees of freedom remain free for drag minimization (for small numbers of loadings, this is reduced to one).

The second limit imposed on the number of pressure constraints is dictated by the number of loadings that are free to influence longitudinal pressure gradient. A maximum of nine loadings does so: loadings 2, 5-8, and 11-14. It has been thought desirable to leave one degree of freedom for drag

minimization for gradient constraints. The number of permissible gradient constraints is reduced by one more whenever a constraint is imposed on C_{mo}, since C_{mo} constraints are satisfied primarily by the same x-dependent loadings used to satisfy gradient criteria.

If the program reaches a limit on the number of pressure constraints, it checks to see if gradient constraints have been imposed. If one or more gradient constraints have been imposed, the program arbitrarily increases the gradient criterion table by 20 percent, and begins anew with no pressure constraints. This process is also cyclic and can be repeated up to 50 times before halting with the solution produced by the last cycle.

If no pressure gradient constraints have been used, the program stops cycling upon reaching either one of the two constraint limits, and retains the last solution.

Unstable solution matrix check. - The cyclic application and removal of constraints just described can result in an "endless loop". I.e., a given constraint will be successively added and then deleted and then added again, due to the LaGrangian multiplier check being incorrect. This condition occurs due to an improper solution matrix formulation caused by imprecise interference drag calculations between loadings. Imprecision in the interference drag calculation is inherent in the panel type approach, and it is not possible to predict when it will result in an endless loop in the optimization. It is possible, however, to analyze each solution matrix for stability, after the matrix is established, by an eigenvalue check.

The test for stability is to calculate the number of zero and negative eigenvalues of the solution matrix. The matrix is unstable if there are any negative eigenvalues or if the number of zero eigenvalues exceeds the number of imposed constraints. This test is performed for each cycle of the optimization. If the test fails, the solution is stopped. It is necessary then for the user to reexamine the loading series and/or constraint application. It has been found that the unstable matrix can usually be corrected by the addition of a Z constraint (e.g., in the case of a wing-fuselage, a Z constraint at 65 percent chord of the side-of-body station is desirable).

Optional pressure level constraint definition. - A study of supersonic wing upper surface pressure constraints has been published by Kulfan in reference 6. The constraint formulation is summarized in figure 3.7-7. It consists of an overall upper surface pressure limit associated with vacuum C_p; an inboard shock pressure limit associated with leading-edge sweepback and fuselage side-of-body slope; and a trailing-edge pressure limit associated with trailing-edge sweepback.

Use of the optional pressure level constraint table is controlled by input AXCPLIM. If used, additional inputs are required to establish basic parameters associated with the pressure coefficient level values, including:

- δ_B = average fuselage growth angle in inboard shock region, + = expanding body
- Λ_{LE} = average wing leading-edge sweepback angle in inboard shock region
- X_R, Y_V = X and Y input values defining inboard shock region

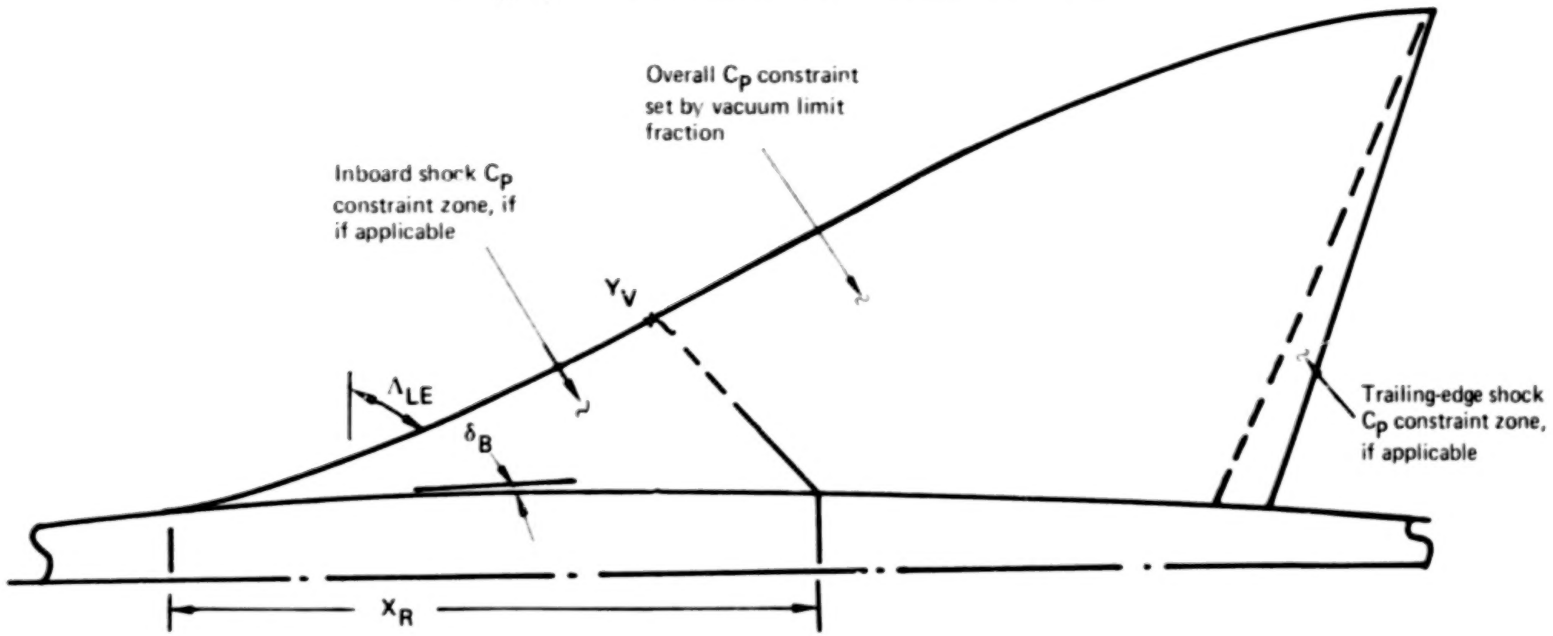


FIGURE 3.7-7.–OPTIONAL C_p LEVEL CONSTRAINT DEFINING REGIONS FOR UPPER WING SURFACE

- Fraction of vacuum C_p for upper surface limit (.70 is a typical value).

- Average inboard wing leading-edge sweepback for inboard shock limit.

- Percent of wing chord at side of fuselage for inboard shock limit to end.

- Percent of wing semispan at which inboard shock region ends. This is typically defined by where the wing leading edge intersects a reverse Mach line from the end of the region at the side of fuselage, as shown in figure 3.7-7.

- Average fuselage area growth angle for inboard shock region. An expanding fuselage has a positive area growth angle which makes the inboard shock limit more severe.

Of these constraints, the inboard shock limit is typically most severe. The program prints out the pressure limits imposed. Using the RESTART capability, as described on page 38, it is readily possible to investigate the sensitivity of the wing design to pressure level constraints through a series of level constraint changes. Since the pressure level constraints (particularly the inboard shock limit) are qualitative in nature, it is recommended that this sensitivity series be run.

Ordinate constraints.- Linear theory produces wrinkles, or kinks, in the computed camber surface aft of wing leading-edge breaks. This is especially noticeable near the wing apex, or at the side-of-fuselage station if there is a fuselage. As a result, Z constraint provisions are provided in the wing design input.

As many as five ordinate constraints can be applied at arbitrary locations on the wing planform, provided that each of the constraints is on a camber calculation line. The program tests the span stations of the ordinate constraints and shifts them to the nearest calculated camber line if they do not lie on one. (This has been done to avoid difficulties with two-dimensional interpolation near the side-of-fuselage).

The program next checks constraint planform locations chordwise. If any constraints are downstream of the trailing edge, they are moved to the trailing edge. If any are ahead of the leading edge, the case is stopped.

When ordinate constraints are used, several cases should probably be run. For the first case, both the maximum number of component loadings being considered and the maximum number of ordinate constraints being considered should be used, and a RESTART deck should be generated. Using the RESTART capability (described on page 38), a reduced number of ordinate constraints may then be employed. Ordinate constraints may be either deleted or switched in order (as is the case with loading numbers). The optimum solution is one in which a minimum number of Z constraints is imposed and which results in a smooth camber surface shape. Cases without ordinate constraints can also be run from a RESTART deck, which includes the component loading ordinate data.

Some thought should be given to the location of ordinate constraints. (In generating the RESTART deck, the value of Z imposed is not important since it can be altered in subsequent runs. The location, X and Y, is important and cannot be altered.) Difficulties possibly requiring ordinate constraints arise at the centerline of a wing-alone solution, at the side of fuselage station, and at wing stations having substantial change of leading edge sweepback angle. One would be tempted, for example, to impose ordinate constraints at, say, 65 percent, 35 percent, and 90 percent chord of the side-of-body wing station, leaving two ordinate constraints free for use further outboard. It would then be feasible, using RESTART, to run a case with different combinations of ordinate constraint at the side-of-body station. An important ordinate constraint has been found to be at 60 to 70 percent of side-of-body station.

In general, ordinate constraints should be used sparingly, since they compromise the number of component loadings available for drag minimization.

Switching the number or order of loadings and ordinate constraints in conjunction with RESTART runs is discussed on page 39.

Solution over-constraint. - There are six types of constraints that can be imposed on the wing design optimization. Constraints include:

(1) Lift coefficient

(2) Pitching moment coefficient at zero lift

(3) Body buoyancy, body upwash, and nacelle buoyancy loadings

(4) Pressure gradient on the wing upper surface

(5) Pressure level on the wing upper surface

(6) Camber (Z) ordinates

It is certainly possible to specify an over-constrained solution -- 5 camber ordinate constraints for a case combining three component loadings, for example. Consequently, a test has been placed in the first part of the optimization program OPTIMUM to detect and correct this situation. The test first sums the number of constraints types (1), (2), (3), and (6) above. If the sum is either greater than or equal to the number of component loadings, the number of ordinate constraints is reduced so that the sum is one less than the number of loadings. If the altered number of ordinate constraints is negative, the program halts; otherwise it proceeds normally. The only exception to this test is the single loading option discussed on page 30, which bypasses the test.

Loading selection. - The loading definitions used in the program are tabulated on page 23, consisting of both analytically defined and configuration - dependent type loadings. These may be input in any order, subject to the condition that the camber-generating version (12-14) of the configuration dependent loadings may not be used without also using the corresponding configuration-dependent loading (15-17). (An exception is the nacelle pressure field loading, discussed on page 26.)

Experience with the wing design program has shown that combinations of the higher order X term loadings (i.e., the X term loadings other than loading 2) tend to produce excessive twist or waviness in the calculated camber surface unless constraints are imposed on wing upper surface pressure level and gradient.

It is always good practice to run the wing alone with the three basic loadings (uniform + linear spanwise + linear chordwise) with only a lift coefficient constraint in addition to any more sophisticated wing design case. Although the three term case cannot accommodate multiple constraints, it serves as a check on the average pressure gradient the wing can be designed to, in addition to providing an approximation to the optimum wing shape.

RESTART option. - A "restart" option has been provided in the program to minimize computer time on runs involving the same planform and Mach number (i.e., different design points in terms of C_L, C_{mo}, ordinate, or pressure constraints). The restart option works as follows: For a given wing planform, Mach number, and set of loadings, most of the computer time is used in calculating the force coefficients and interference coefficients associated with all the component loadings. The calculations involving the solution of an optimum combination of loadings, with or without constraints, are relatively quick. However, it may be desirable to look at a number of different optimization or constraint solutions. Therefore, on successive cases involving the same basic loadings, it is possible to bypass the component loadings solution and go directly to the optimization routines. This is done by setting RESTART= -1. in the program input for cases 2 and on.

Alternatively, if the program cases are to be input at a later time, the component loadings data may be punched into cards (using RESTART = 1.) and read back in to the computer through use of RESTART = 2. The RESTART = 2. data deck includes, as well, the definition of any configuration-dependent loadings that were present in the wing design program at the time the data deck was punched.

RESTART = 3.0 makes use of restart data written (when created) on a disk file (TAPE3) which may later be reread in the wing design program. For re-entry into the wing design program after a lift analysis program run, for instance, RESTART = 3.0 functions the same as RESTART = -1. (RESTART = -1. may be used only for successive wing design cases without exiting the wing design program.) If TAPE3 is saved after the RESTART data are generated, the saved file may be used instead of the punched RESTART deck. (To do this, set RESTART = 3.0 and substitute the saved file name as the eighth argument in the

program call card file list. The saved file may not be used in place of the punched RESTART deck because of format differences. A summary of file usage on the program header card is given in volume 3, computer program description.)

The restart option also will work in the case of a decreased number of loadings. E.g., if a maximum (17) loading case were run, then the force and interference loading terms for all lesser combinations of loadings are available. Successive cases can then be run with different loading combinations to check the design sensitivity to certain loadings, without repeating the basic loadings calculations. Any combination of loadings involving the set used when the RESTART data were generated may be employed. The number and the order of the loadings may be altered as desired.

If the RESTART data deck is used, it is not necessary to recalculate any configuration-dependent data (since these are preserved along with the basic wing loading data). With respect to the wing design solution that was possible at the time the RESTART data were generated, the RESTART deck may be used for any wing design having:

1) The same or fewer loadings (order may vary)

2) Same fuselage geometry, angle of attack, and side-of-fuselage station.

3) Any C_l, C_{mo}, or pressure constraint condition.

4) The same or fewer ordinate constraints (order may vary). The value of Z at these locations can be changed.

The procedure for altering the order of Z constraint data is somewhat awkward (but leads to considerable simplification in the computer code for arbitrary loading and Z combinations). In the original calculation of the RESTART data, the ordinate constraint locations are numbered consecutively from 1 to ANZ, as input. In a RESTART run, the same (or fewer) Z constraint locations may be selected in the same order (ANZ = number of locations) and the corresponding values of Z may be changed if desired. If the order of Z (and number of constraints also, if desired) is to be changed, the number of Z constraints is input as negative. The next card then identifies the order relative to the RESTART generation order. The next card then gives the corresponding Z values. An example illustrates the procedure:

RESTART generation input deck, 5 Z constraints:

1.	1.	1.	2.	5.	card 7
X_1	X_2	X_3	X_4	X_5	card 7A
Y_1	Y_2	Y_3	Y_4	Y_5	card 7B
-10.	-8.	-6.	-7.	-3.	card 7C

Subsequent RESTART run, 5 Z constraints, ordered 5-4-3-2-1 with respect to the RESTART generation run:

1.	1.	1.	2.	-5.	card 7
5.	4.	3.	2.	1.	card 7D
-3.	-7.	-6.	-8.	-10.	card 7E

The values of Z were not changed to help clarify the example. Notice that the values of X and Y were not input; they are obtained from the RESTART data corresponding to the original order number. Location 1. on card 7D identifies X_1, Y_1, etc.

If a lesser number of Z constraints is desired, only the ones wanted need to be identified, e.g.:

1.	1.	1.	2.	-1.	card 7
4.					card 7D
-7.					card 7E

Planform considerations and spanwise integration. - The wing design program is a direct type solution; i.e., a wing shape is calculated from a known pressure distribution. It is not necessary to calculate the wing shape at all spanwise stations in the grid system used to represent the wing; only a representative set of spanwise stations is used. The lift, drag, and pitching moment coefficients are then computed from a spanwise integration of the characteristics obtained at the selected spanwise stations.

In the program input, the camber surface calculations are performed at a standard set of 23 semispan stations unless otherwise specified. If the planform is irregular, particularly along the leading edge, additional spanwise stations in the vicinity of these irregularities should be input to improve the solution accuracy. (This is done through inputs TJBYMX and TJBYS, as described in Section 4.)

In addition, it has been found that, for a wing alone case, the wing root singularity and the corresponding root camber line can often be moderated by substituting a parabolic apex for the sharp apex common to supersonic wing planforms. This will be performed automatically in the program if the input YSNOOT is not zero. The program then fits a parabola tangent to the wing leading edge at YSNOOT, with symmetry about Y=0.

Because the computed camber surface slopes tend to exhibit some irregularity near the leading edge (due to the sawtooth nature of the grid system), a smoothing option is provided in the program. This is activated by the code SMOOTH in the program input. The smoothing technique involves averaging the computed surface slopes of each grid element with the slopes of adjacent elements, which suppresses any erratic slopes of individual elements.

Lift Analysis

Given a wing planform, camber shape, and Mach number, the lift analysis program solves for the lifting pressure distribution and force coefficients for a range of angles of attack. As options, the program will also include the effects of:

- Fuselage (circular or arbitrary cross-section, arbitrary camber and incidence)
- Nacelles
- Canard and/or horizontal tail
- Wing trailing edge flaps and/or incremental wing twist

Fuselage solutions. - Fuselage effects are obtained by calculating the isolated fuselage upwash field, then calculating the wing solution in the presence of the fuselage upwash field, then recalculating the fuselage forces in the wing flow field, and combining the solutions by superposition.

Two options are available for calculating the fuselage data:

- A slender body solution for a fuselage which is circular (or nearly so) in cross-section. Fuselage must have zero area at nose, and cannot have significant discontinuities in longitudinal area distribution.
- An arbitrary fuselage cross section solution. Fuselage area need not be zero at nose, and fuselage can have discontinuities in either area or first derivative of longitudinal area distribution.

Fuselage of circular cross-section. - For the fuselage of circular cross-section, two options are provided to simplify input. One is a wing-fuselage intersection option. This feature tracks each wing percent chord line out through the side of the fuselage (circular in cross-section), and breaks the wing solution into the proper exposed and carry-over type lifting pressure calculations. Alternatively, the side-of-fuselage span station may be input, either as a constant or as a table of values to override the wing-fuselage intersection option.

The local fuselage upwash angle is strongly affected by span station and wing height on the side of the fuselage. The side-of-fuselage span station must be carefully input to avoid exposing any wing area to the upwash field that is actually inside the fuselage.

The circular cross-section fuselage solution also contains an option to calculate the buoyancy field due to unequal fuselage area growth above and below the wing. This pressure distribution, termed asymmetric fuselage buoyancy, is calculated by splitting the fuselage area into pieces above and below the wing and adding the resultant area growth onto the fuselage forebody area distribution. (The fuselage is again considered circular, and the side-of-fuselage Z value is used to define the above-wing and below-wing area pieces). The asymmetric fuselage term is zero, of course, in the case of a mid-wing arrangement. The asymmetric buoyancy calculation is requested by input SYMM (value greater than zero).

Fuselage of arbitrary cross-section. - The arbitrary fuselage cross-section option makes use of the digitized cross-section definition (see fuselage geometry input, page 64). For a specified set of X stations, the cross section is defined by sets of Y and Z values.

If the arbitrary fuselage option is selected, additional input is required (on card 28, page 104). The additional input defines print codes for solution output and discontinuity codes.

The basic geometry of the program provides for four fuselage segments. Discontinuities in fuselage area or first derivative can be accommodated only at the first fuselage station of a segment. A discontinuity in first derivative is assumed at the beginning of the first segment (i.e., at fuselage nose). At the beginning of succeeding segments (if used), a discontinuity code is required. If the code is 0., no discontinuity is requested, and the succeeding segment is considered to be a smooth extension of the previous segment. If the discontinuity code is 1.0, a discontinuity is inserted, and the fuselage geometry effectively restarts at the beginning of the succeeding segment. Thus, up to three discontinuities in either area or first derivative may be requested.

Within a segment (or across segments if no discontinuity is encountered), the program fairs longitudinally between like points in Y-Z pairs; i.e., the ith point of a cross section is connected longitudinally to the ith point of other X stations. If there is no discontinuity between segments, the same number of Y-Z pairs must be used for the continuous segments. In creating the fuselage input, the definition should first be laid out for the most complex X station, and other stations then defined through consideration of the longitudinal fairing connections. Duplicate points at a given X station (i.e., two or more sets of Y-Z pairs identical) are not allowed, except at the fuselage nose or tail if area = 0.

If there is a discontinuity between segments, the number of defining Y-Z pairs between the discontinuous segments may be changed, if desired.

There are two parts to the arbitrary fuselage cross-section solution: an equivalent axisymmetric body (thickness) solution and an arbitrary shape cross-flow solution. The program solves for the area and centroid of all cross sections from the digitized input. The axisymmetric solution employs

the area distribution disposed along a straight axis. The actual contour shape and centroid are used in the cross-flow solution, with the centroid used as the local reference point in the process of transforming the given cross-sectional shape into a circle. The cross-section contour as seen from its centroid must be simply connected and describable by a single valued function in polar coordinates.

There is another important point with respect to the arbitrary fuselage solution. Fuselage upwash data in the plane of wing/canard/tail is calculated normal to the fuselage centerline at the X stations of the cross section definitions. In the subsequent lifting solutions, these upwash data are interpolated as required over the planforms of the lifting surfaces. It is necessary, therefore, to provide sufficient fuselage cross-sections to allow reasonable accuracy in the interpolation. Also, surface pressure coefficients are calculated at the fuselage input geometry points. It is necessary to have a good and sufficient distribution of defining points to permit an accurate integration of these pressures over the surface for lift, drag, and pitching moment.

The computer-generated wing-fuselage intersection and asymmetric fuselage buoyancy simplification options used with the circular cross-section solution are not applicable to the arbitrary fuselage case. The wing-fuselage intersection (code ANYBOD = -10.) and asymmetric fuselage area distributions (if desired, code SYMM = 2.0) must be input. Both of these options require input of the data at the same percent chords used in the camber surface definition. (Exception, constant y intersection may be specified, ANYBOD = Y.)

Arbitrary fuselage force coefficients. - Surface pressure coefficients are integrated over the fuselage area for a series of angles of attack to obtain fuselage lift, drag, and pitching moment. The drag thus obtained includes the thickness wave drag of the fuselage. At zero angle of attack (or nominally so), this drag may be compared to the fuselage wave drag from the near-field or far-field programs.

For drag bookkeeping purposes, the drag-due-to-lift force summaries include the drag of the fuselage lift distribution acting on the fuselage camberline shape. In the case of the arbitrary fuselage solution, the camberline is taken as the plot of centroid Z value of the input cross-sections versus fuselage station. This definition produces exact correspondence between the arbitrary fuselage and circular slender body options for the case of a fuselage having circular cross-sections.

Wing leading-edge thrust. - The basic lifting solution for the wing does not include the theoretical leading-edge thrust term. The leading-edge thrust calculation is requested by an input code (XESUC).

The theoretical leading-edge thrust calculations are performed for a series of angles of attack, from -4^{0} to $+ 6^{0}$ with respect to the basic input angle of attack. The sectional thrust values (based on average chord) are

summarized for representative angles of attack and integrated to get the wing total thrust coefficient. The configuration force coefficient data are then calculated and summarized for:

- No leading-edge thrust
- Full leading-edge thrust assuming attached flow, with line of action of the local thrust vector tangent to camberline at nose of airfoil
- Separated flow according to the Polhamus assumption, whereby the theoretical leading-edge thrust is rotated 90°, with line of action of the local thrust vector normal to camberline at nose of airfoil.

There is a further "attainable thrust" calculation which is derived from the theoretical leading-edge thrust and is requested by another input code (ATTLE). The attainable thrust solution requires additional input involving Reynolds number and airfoil characteristics.

The attainable thrust solution computes a factor, K_T, which relates the attached flow portion of the leading-edge thrust value to the total leading-edge thrust. When K_T is less than 1.0, leading-edge thrust is not completely of the attached flow type, and the thrust vector is rotated enough to define an attached flow component equal to K_T times the theoretical leading-edge suction. When K_T = 0., separated flow of the Polhamus type occurs (i.e., sharp leading-edge case).

The attainable thrust K_T factors are calculated and printed for the same span stations and same series of angles of attack as the theoretical leading-edge thrust solution. The configuration force coefficient summaries then list also the lift and drag coefficients corresponding to the attainable thrust solution.

Typical leading-edge thrust values for a representative wing are presented in figure 3.7-8.

Nacelles. - The nacelle calculations are very similar to the solution used in the near-field wave drag program. The pressure fields imposed by the nacelles on the wing, and wing-on-nacelles, are computed and their combined effect on the lifting solution obtained through superposition. The effect of the nacelles on the wing drag-due-to-lift can be substantial because of lift contributed by the nacelle pressure field, Both "wrap" and "glance" solutions for the nacelle pressure field are available, as described in Section 3.6.

Canard and horizontal tail. - Canard and horizontal tail lifting pressure distributions and force coefficients are calculated as for the wing case. The program assumes that a canard is located forward of the wing and a horizontal tail aft of the wing. The effects of downwash from upstream lifting surfaces (if any) are included in the solution. Leading-edge thrust is not calculated for these surfaces.

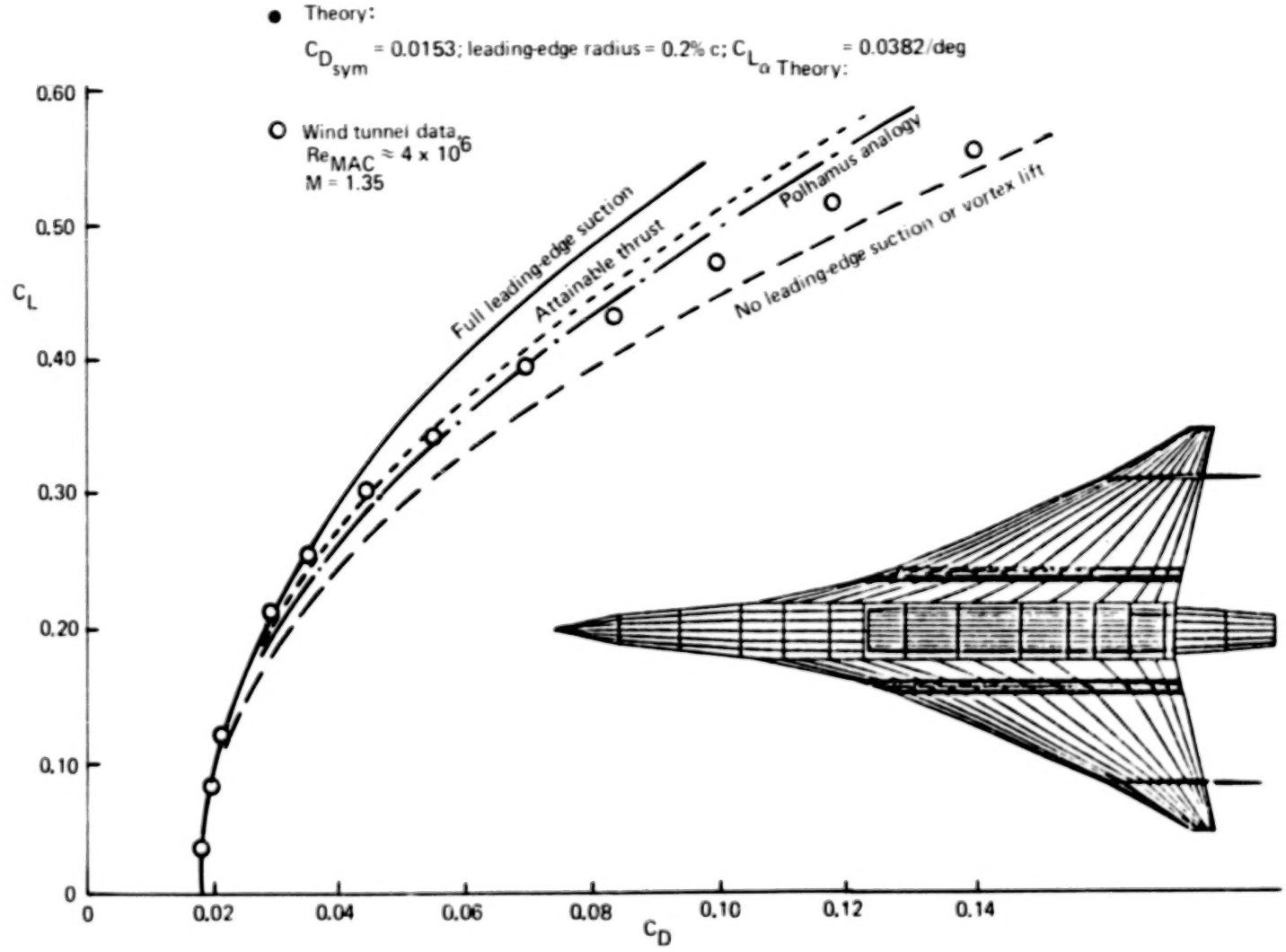


FIGURE 3.7-8.—LEADING-EDGE SUCTION EFFECTS, M = 1.35

Downwash "shift" options. - The basic theoretical solution assumes downwash from the canard or wing propagates directly aft. Since the downwash in the real flow case must follow the fuselage contour, a shift feature in the program translates the downwash field laterally to account for fuselage radii change between a generating (canard or wing) and affected (wing or tail lifting surface. The downwash shift can have an appreciable effect on the calculated characteristics, as shown in figure 3.7-9.

The shift feature is controlled by input codes. If the control codes are left blank, the downwash will be propagated directly aft. Alternatively, the downwash can be either shifted according to the side-of-fuselage Y values of canard, wing, or tail or shifted a specified amount, as described in the data input section (4.0).

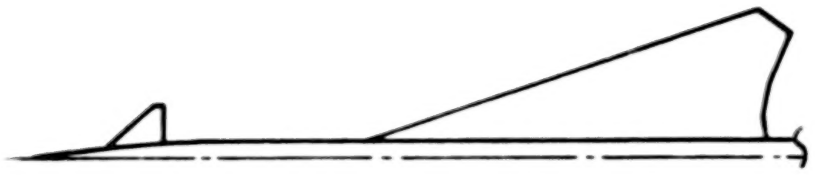
Experimental comparisons. - Theoretical calculations for a typical supersonic configuration are compared with corresponding wind tunnel data in figures 3.7-10 and 3.7-11 (wing-fuselage-nacelles) and figures 3.7-12 and 3.7-13 (incremental effects of horizontal tail). The theoretical buildup of the zero-lift drag coefficient is given in figure 3.7-10.

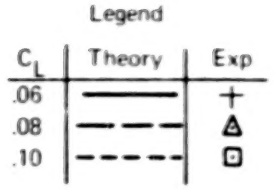
The lift analysis program contains an optional pressure limiting feature for the wing surface pressures which operates somewhat differently from the one in the design program. In the design case, the local wing angle of attack is not allowed to exceed the value associated with a pressure limit condition. In the analysis case, the pressure coefficient limit is imposed, but the local wing incidence may greatly exceed the value at which a limit is first encountered.

When the pressure limiting option is used, a set of configuration angles of attack for the solution must be provided, and the configuration thickness pressures from the near-field program must be provided to permit limiting of the total surface pressure. A solution for a typical wing through an angle of attack series using the pressure limiting feature is shown in figures 3.7-14 and 3.7-15. The limiting feature greatly improves the linear theory representation of the wing pressure distribution as angle of attack is increased.

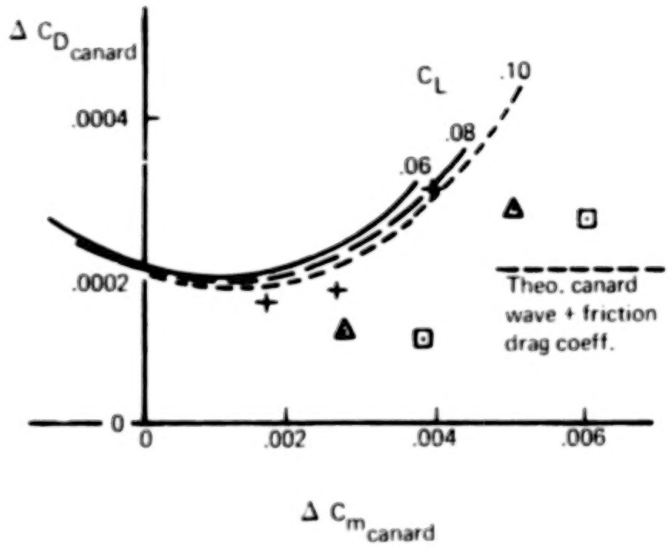
Configuration-dependent loadings. - One mode of lift analysis program usage is to generate configuration-dependent data for the wing design program. These data are produced as follows:

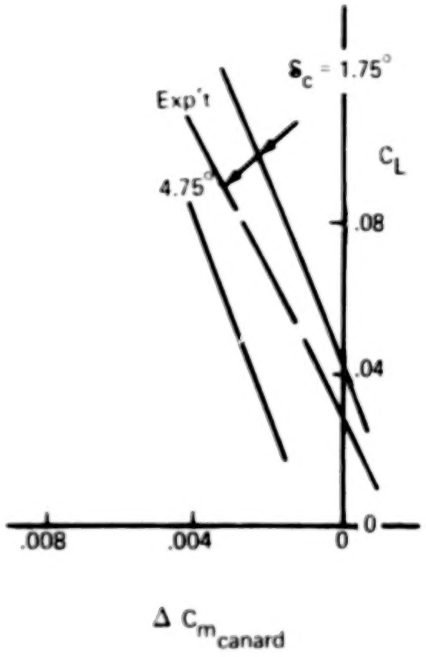
Data	Description	Requirements
Nacelle pressure field	Pressure field caused by nacelles on wing.	Call for nacelles (AJ3 = 1.0)
Fuselage upwash field	Pressure field induced on wing by fuselage upwash.	Calculate fuselage effects on wing
Fuselage buoyancy field	Pressure field induced on wing by unequal fuselage volume above and below wing.	SYMM ≠ 0.



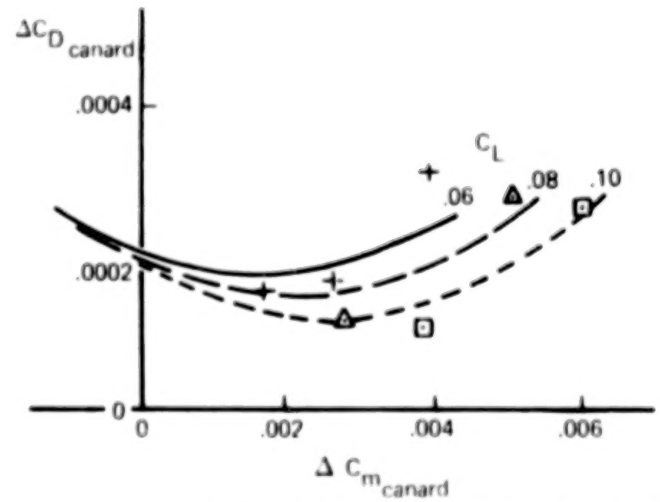


NO CANARD DOWNWASH SHIFT





WITH CANARD DOWNWASH SHIFT



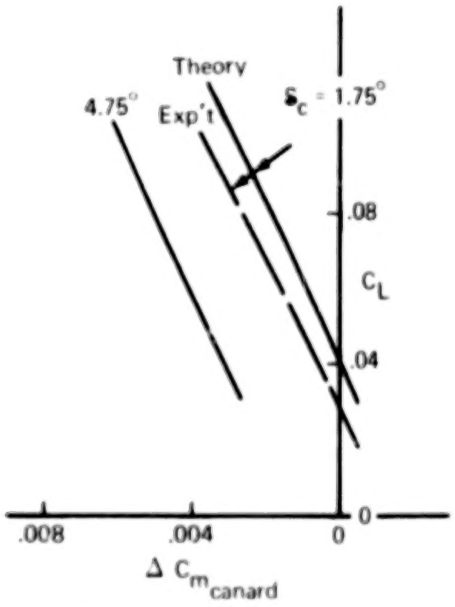


FIGURE 3.7-9.—EFFECT OF CANARD DOWNWASH SHIFT ≈ M = 2.7

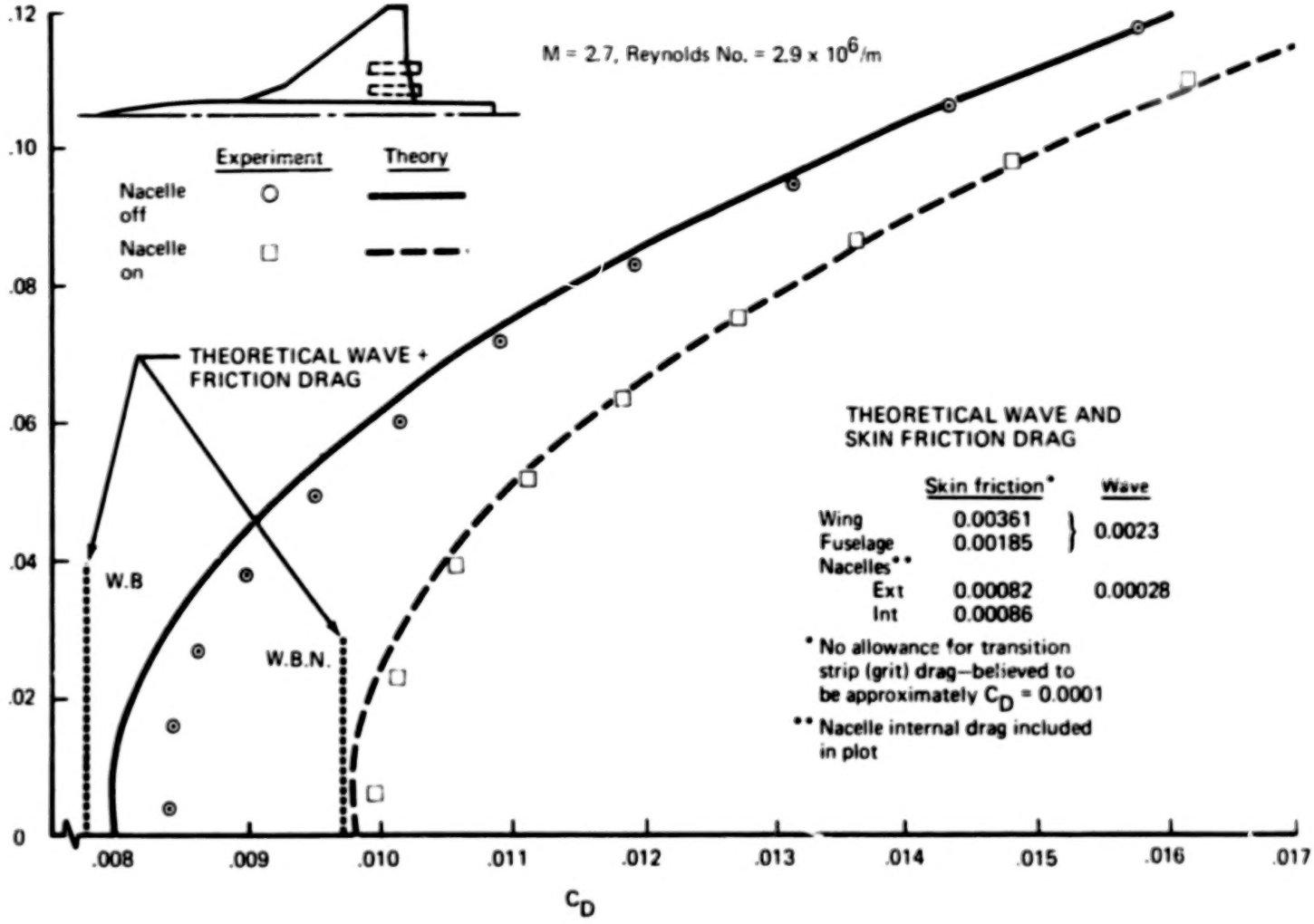


FIGURE 3.7-10.—DRAG POLAR COMPARISON

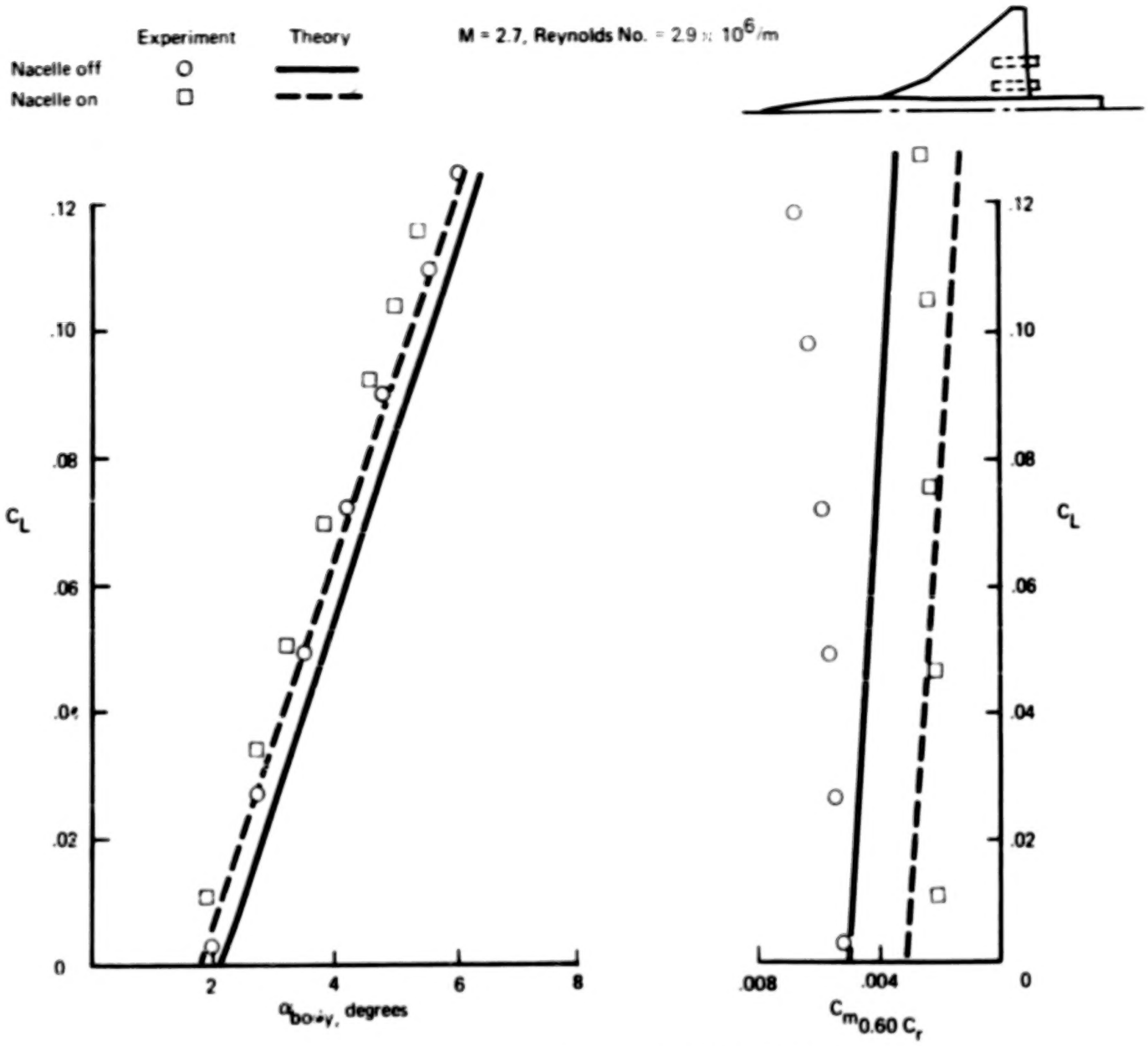
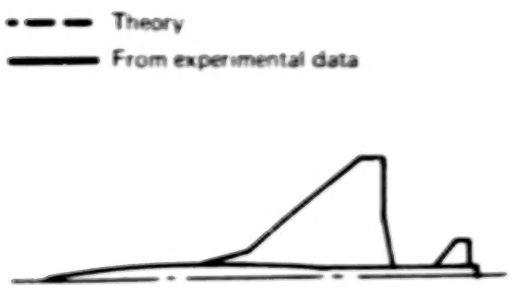
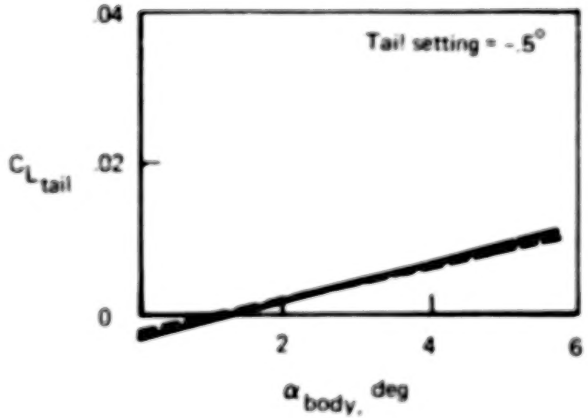


FIGURE 3.7-11.—FORCE COEFFICIENT COMPARISON



M = 2.7

Reynolds No. = 2.9×10^6/m



Note: Horizontal tail setting referred to wl. Angle relative to wing z = 0 plane is 1.25° larger.

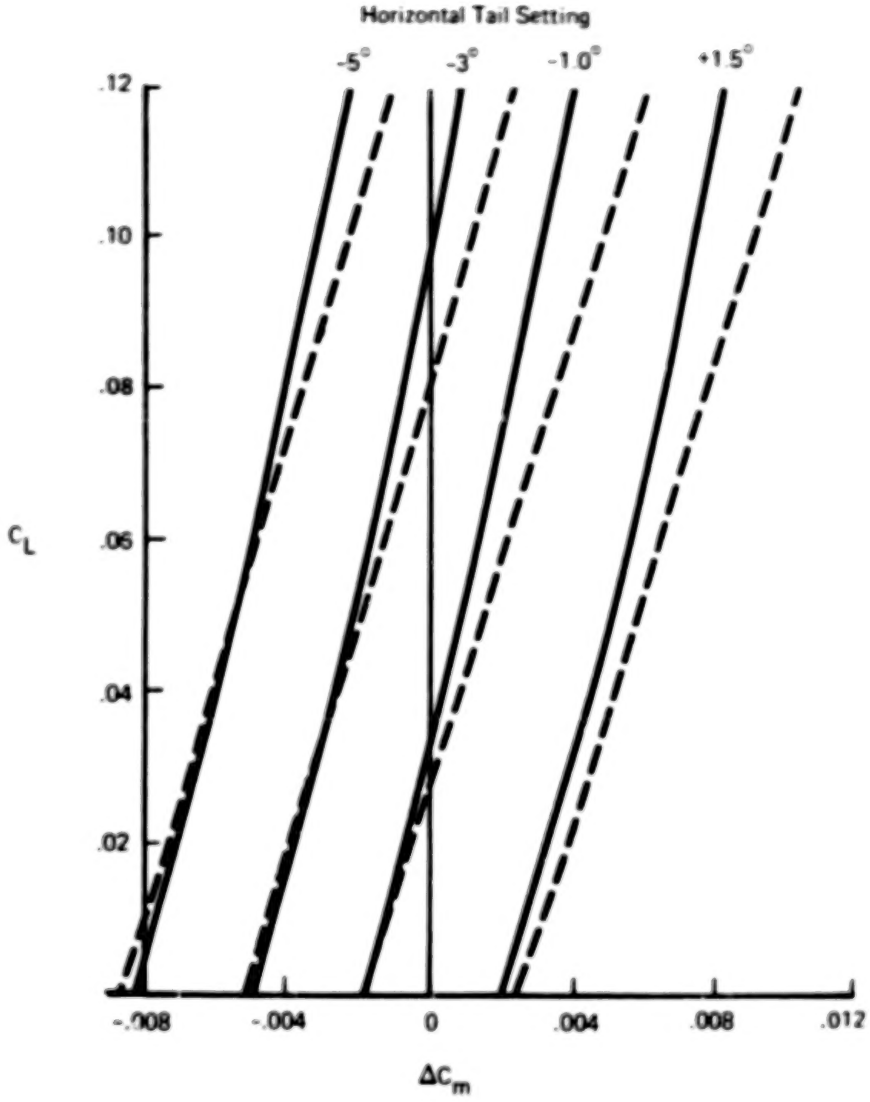


FIGURE 3.7-12 – HORIZONTAL TAIL EFFECTS

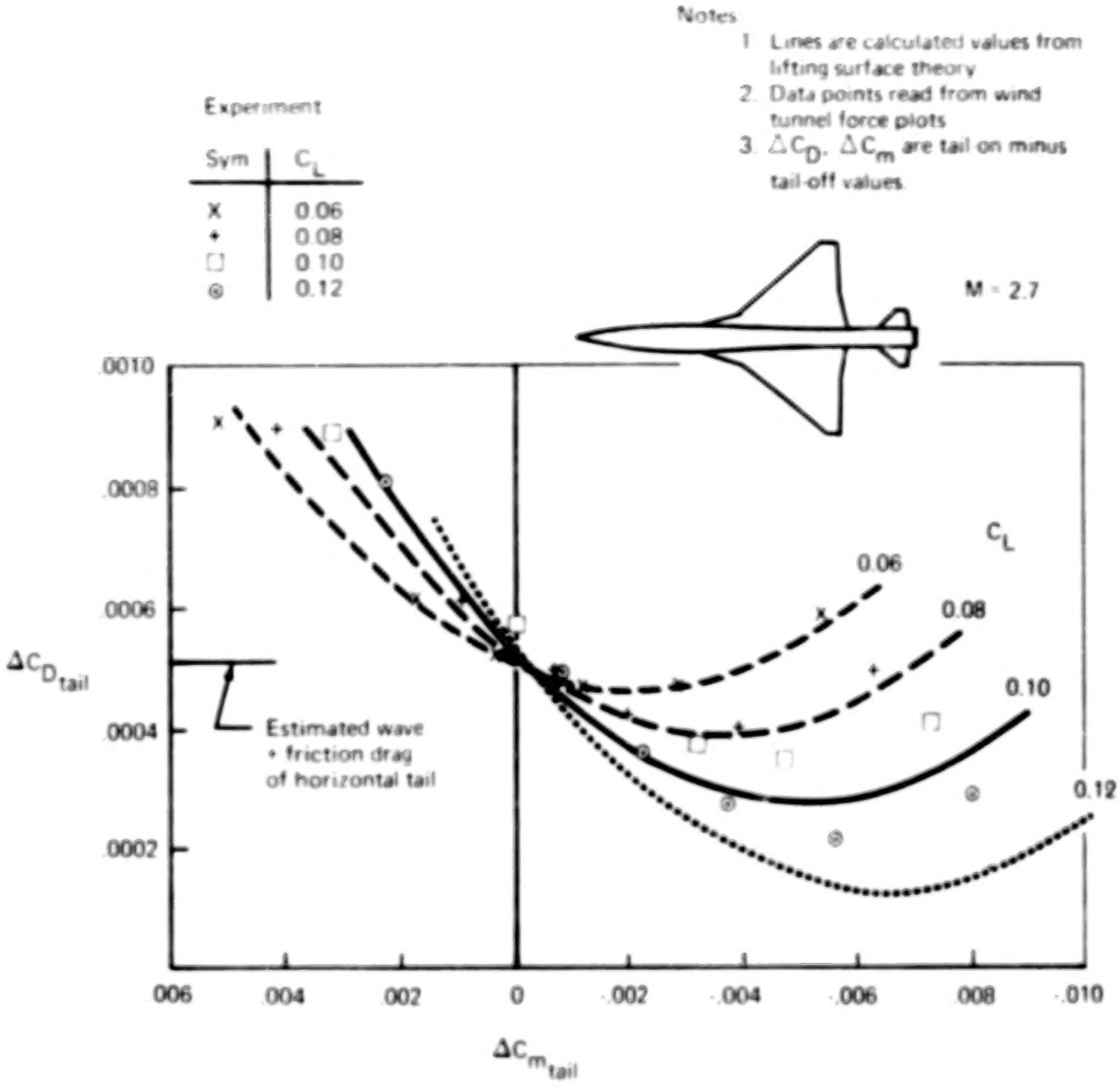


FIGURE 3.7-13.–HORIZONTAL TAIL EFFECTS

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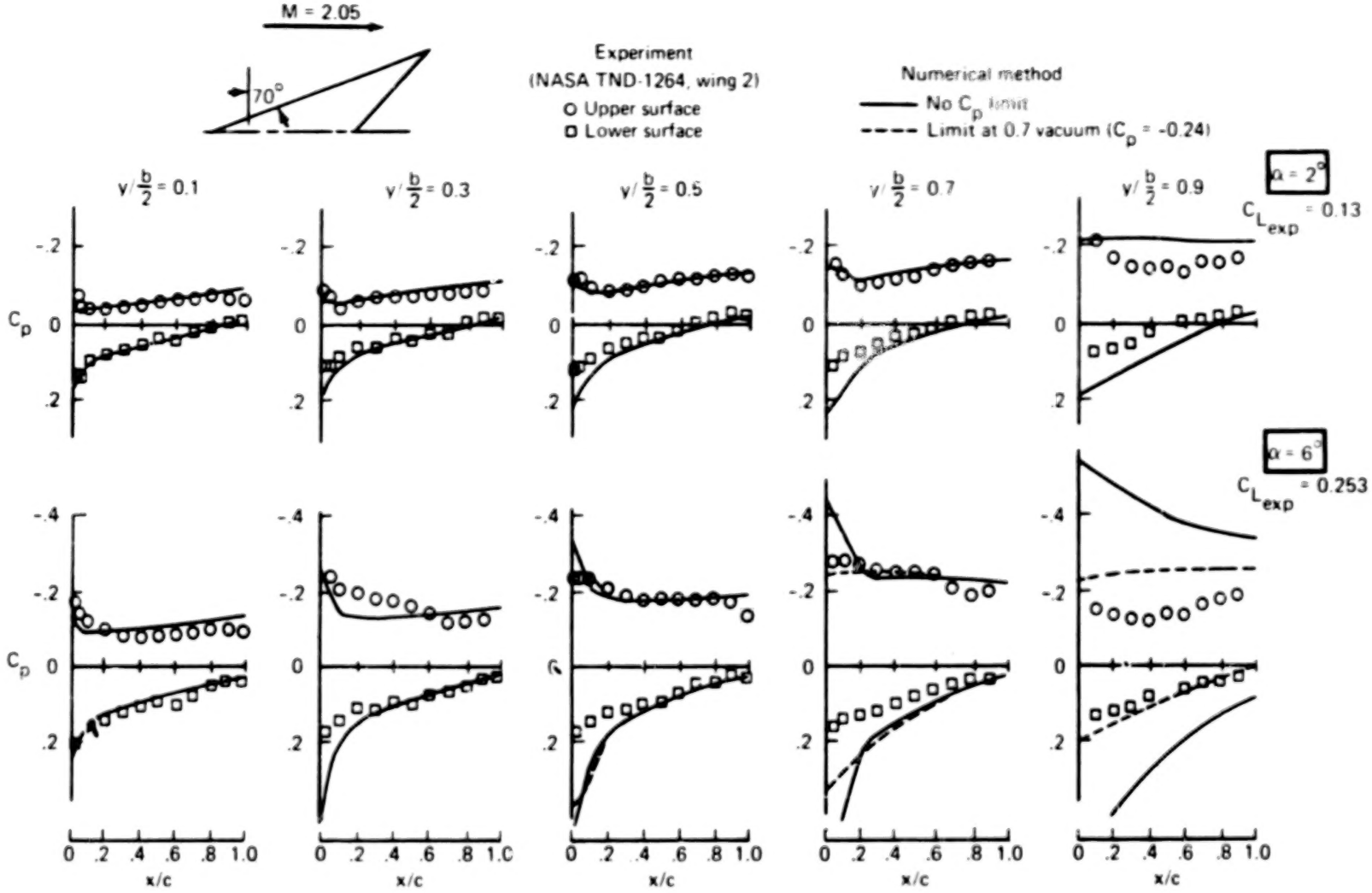

FIGURE 3.7-14.—PRESSURE COEFFICIENT COMPARISON—WING 2 (TWISTED AND CAMBERED WING, C_L = 0.08)

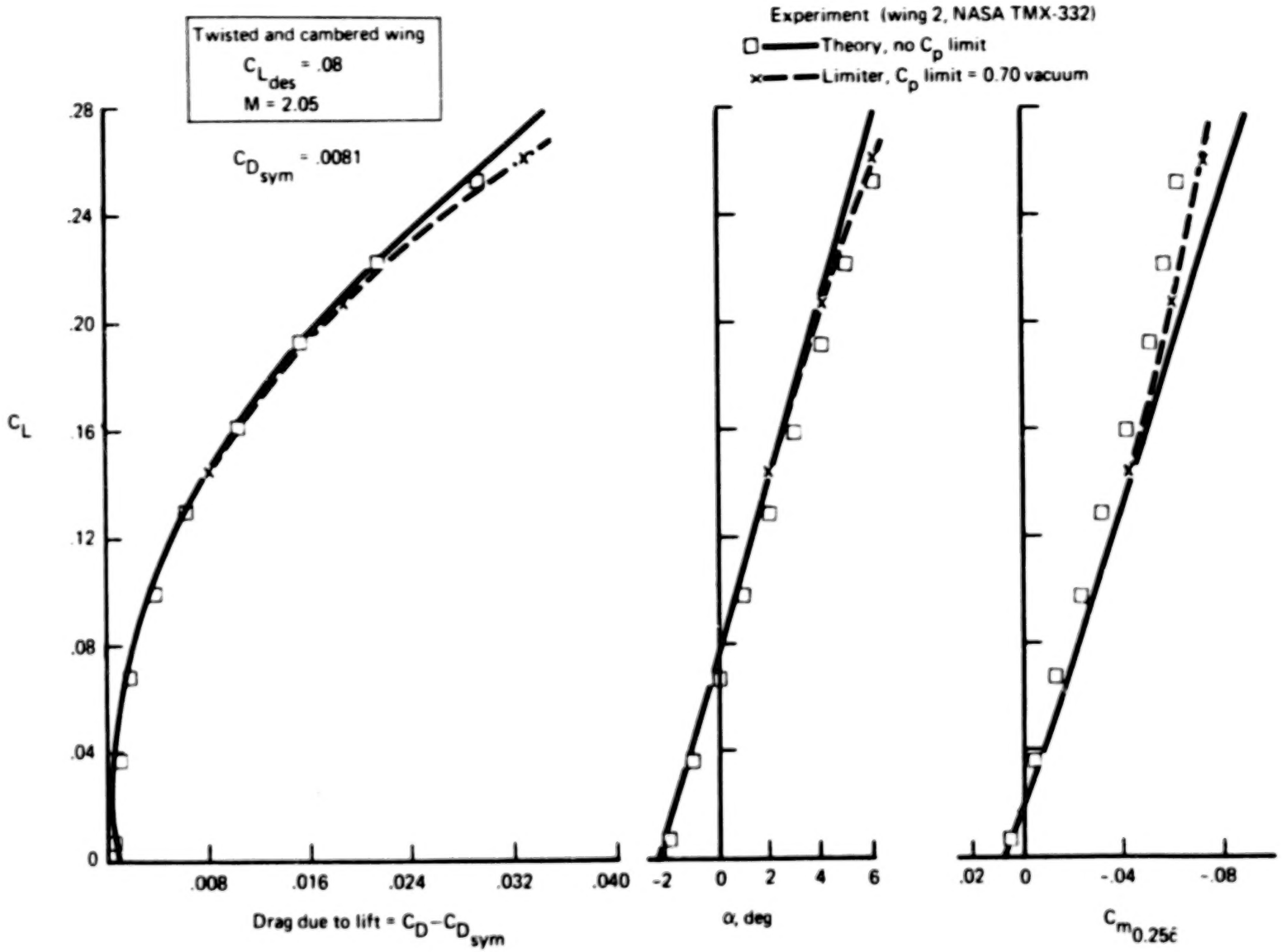

FIGURE 3.7-15.–TEST-THEORY COMPARISON–WING 2 ($C_{L_{des}}$ = 0.08)

Upon execution, the program then loads the pressure fields into the proper system common blocks.

If the fuselage buoyancy field is not requested (i.e., SYMM = 0.), the program computes the pressure field due to a mid-wing arrangement. This is done so that a thickness pressure field due to the fuselage will be available for pressure limiting calculations, if desired.

In calculating the fuselage upwash or buoyancy fields, it is important to recall the influence of wing height on the side of the fuselage. This strongly affects both the local upwash angles, and the above-and-below wing area distributions.

Calculation of the fuselage upwash field may be done in either of two ways; the principal condition is that the resultant pressure field is that due to upwash only. In the computer program, this is handled by inputting a camber surface having approximately the correct wing-fuselage relationship (wing height, etc.), but then zeroing the wing slopes in the camber surface calculations (by setting WHUP = 1.0). In iterative cycles, the wing camber surface and fuselage relationship can be refined.

3.8 Wing Pressure Summary

The wing pressure program summarizes the wing upper surface and lower surface pressure distributions at input angles of attack or lift coefficients. Pressures from the fuselage, nacelles, wing thickness, and wing lift are superimposed in any desired combination and printed. There are several options for defining the location of pressure coefficient values, as described in the input description (page 106).

The wing pressure summaries are saved on TAPE19 for offline plotting. The basic component pressure fields are also saved on TAPE20, if requested, in a format compatible with the wing design program configuration-dependent pressure input. Format of the data on TAPE19 and TAPE20 is as follows:

TAPE19 (Pressures at input condition)

Format	Data
A24,I4	Planform identification, number of planform stations
F6.4, 4(3X,F10.5)	Semispan fraction, Y, X_{LE}, X_{TE}, chord
7A10, I2	Case identification, NCOND (number of C_L or α values)
2(6X, F10.5), I4	α(deg), C_L, NAYPC (number of Y stations for data)

There follow NAYPC sets of data, first the Y station and number of X stations (NAXPC) at this Y station; then NAXPC sets of X and C_p values.

10X, F10.5, I6	Semispan fraction, NAXPC
3(5X, F10.5)	Chord fraction, $C_{p\ upper}$, $C_{p\ lower}$

There follows another α, C_L, NAYPC data set and subsequent pressure data until NCOND conditions have been written. TAPE19 is rewound in the geometry interface program for the wing pressure summary module.

TAPE20 (Basic component pressure fields)

Pressure coefficients for the configuration-dependent loadings in a format compatible with the wing design program are written on TAPE20 according to the following format. The first record of each data set describes the data set to follow.

Format	Data
5X, A27, 2(3X,I4)	Data description, NX (number of X stations), NY (number of Y stations)
10F7.3	X station array corresponding to NX
10F7.3	Y station array corresponding to NY

There follow NY sets of pressure data, with C_p values corresponding to NX.

Each NY set begins a new record.

10F7.3	C_p values

The above format is used for the wing lifting pressures at basic angle of attack and for the wing thickness pressures. The same format is used for the fuselage pressure field acting on the wing except that two sets of C_p values are listed; first, the complete set for pressures acting on the wing upper surface and second, the complete set for lower surface pressures.

The nacelle pressure field uses a similar format, modified by the provision for a different set of X values at each Y station. The pressure field written is that for nacelles below the wing.

Format	Data
5X, A27, 2(3X,I4)	Data description, NX, NY
10F7.3	X station array for each Y station (NY sets)
10F7.3	Y station array coresponding to NY
10F7.3	C_p values for each Y station, in same order as X station arrays (NY sets)

The component pressure data are written on TAPE20 if requested by the input code DPRINT. TAPE20 is rewound in the geometry interface program for the wing pressure summary program.

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4.0 INPUT FORMAT

Input requirements for the system are given in this section and consist of:

- Executive control card summary
- Basic geometry definition
- Additional data input for programs of system

The usual input format is 10 field - 7 digit, punched with decimals to the left in the card fields. Some data (particularly the control codes in the basic geometry) are input in integer form, without decimal, to the right in the card field. The formats are identified in all cases.

To provide design or analysis flexibility, there are numerous program options that are controlled by input codes. Where there is a "normal" way of handling the option, the code is defaulted to zero (i.e., if the field contains a zero or is blank, the "normal" solution will be calculated).

****NOTE****

The allowable input number sizes are:

- No more than 5 digits to left of decimal
- No more than 4 digits to right of decimal

4.1 Executive Control Card Summary

Configuration input and program execution are ordered by means of control cards read at the executive level.

The control cards consist of a few alphanumeric characters starting in column 1.

Geometry input. - The configuration geometry is read and manipulated in the geometry module. Geometry may be input as all-new, or as a replacement or addition to existing geometry. The control cards for geometry input are:

GEOM NEW — All-new configuration description follows, and any previous geometry is purged. (Leave one column space between GEOM and NEW).

GEOM — Input geometry is added to (or replaces) existing description.

Geometry update. - The basic geometry description contained in the geometry module may be updated using data contained in 0, 0 level common blocks. This applies to a new fuselage definition (i.e., optimized fuselage from the far-field wave drag program) or a new wing camber surface definition. The control cards are:

FSUP	Fuselage will be updated to definition contained in /OPBOD/. The /OPBOD/ definition is created each time the far-field wave drag program executes the optimum-fuselage-with-restraints case.

If the fuselage update is requested, a second card, telling how to perform the update, is required. Punch (starting in column 1) the following code:

-1.	Fuselage is to be redefined at same x stations as previous definition.
1.	Fuselage is to be defined at 50 equally spaced stations.
WGUP	Wing camber surface will be updated to the definition contained in /CAMBER/. The /CAMBER/ definition is created each time the wing design program executes and produces a camber surface for a specified set of conditions.

The user must remember that the update for fuselage or camber surface will require that the /OPBOD/ or /CAMBER/ definition be current. These common blocks will contain the last definition produced by the far-field wave drag or wing design programs.

Program execution. - Execution of the programs in the system is ordered by the following cards:

PLOT	plot program
SKFR	skin friction program
FFWD	far-field wave drag program
NFWD	near-field wave drag program
ANLZ	lift analysis program
WDEZ	wing design program
WPLT	wing pressure summary program

The control card for program execution is the first card of the set describing the program data input. Individual program inputs are given on the following pages.

Multiple case execution with the basic programs of the system is possible, as in the stand-alone versions of the programs. The data for successive cases are stacked as described in the program input description. At the end of the data stack, an END card is required to terminate the program. The END card is not used for the geometry input, however.

Interactive graphics. - The graphics subroutines in the system are activated by the executive card CRT (punched in first three card columns). The CRT card may be placed anywhere in the data deck that an executive card may be read. If no CRT card is included, the system will execute without accessing any of the graphics programs.

A description of the interactive graphics part of the design and analysis system is presented in the Appendix.

4.2 Geometry Program

The geometry program stores the basic geometry data, and stacks it as required by the individual programs of the system.

Access to the geometry program, to store or alter the configuration description, is through the GEOM or GEOM NEW control card (see executive control card summary).

The format of the geometry input uses both integer (control cards) and floating point numbers. All integers are punched right justified in their fields on the cards, without decimals. All floating point numbers are punched, with decimals, to the left of the field in 10 field, -7 digit format. The program logic uses the component control codes (J1, J2, etc.) on card 3 as follows:

Value	Use
0	Component will not be input. However, if the component has previously been input (and not purged by a GEOM-NEW card), the 0 is interpreted as a 2.
2	Previously input component is left as is.
Other	New input for this component replaces previous input.

The logic of treating a 0 as a 2 for existing components is to protect data on the geometry file from inadvertent loss. Then, if it is desired to add or change a configuration component on successive runs, only the new component need be addressed.

A control code other than 0 or 2 instructs the program to completely replace the previous component description with a new one. It is not possible to add a fin or nacelle to a previous fin or nacelle; the new description must be complete in itself.

Deletion of a component is possible only through purging the entire configuration, using the GEOM NEW card.

Card Number	Card Column	Decimal Required	Variable Name	Description
1				GEOM or GEOM NEW
	1-4			GEOM = geometry addition
	1-8			GEOM NEW = all new geometry
2	1-70			Any desired title information.
3	1-3	NO	J0	Reference geometry code.
				0 = Reference geometry not required (plot program)
				1 = Read reference area, $\bar{c}$, x_{cg}
				2 = Reference geometry same as previous case.
3	4-6	NO	J1	Wing input code
				-1 = Read uncambered wing
				0 = No wing
				1 = Read cambered wing
				2 = Wing same as previous case.
3	7-9	NO	J2	Fuselage input code
				-1 = Read circular fuselage
				0 = No fuselage
				1 = Read arbitrarily shaped (digitized) fuselage
				2 = Fuselage same as previous case
				3 = Read circular fuselage and perimeter values.
3	10-12	NO	J3	Nacelle input code
				0 = No nacelles
				1 = Read nacelles
				2 = Nacelles same as previous case.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	13-15	NO	J4	Fin input code 0 = No fin 1 = Read fin data 2 = Fin data same as previous case.
3	16-18	NO	J5	Canard (or horizontal tail) input code 0 = No canards 1 = Read canard data 2 = Canards same as previous case.
3	19-21	NO	J6	Fuselage simplification code -1 = Uncambered circular fuselage 0 = Cambered circular or arbitrary fuselage. 2 = Complete configuration is symmetrical with respect to X-Y plane, which implies uncambered circular fuselage if there is a fuselage.
3	22-24	NO	NWAF	Number of airfoils describing wing. $2 \leq NWAF \leq 20$.
3	25-27	NO	NWAFOR	Number of ordinates defining each airfoil section. $3 \leq NWAFOR \leq 20$
3	28-30	NO	NFUS	Number of fuselage segments $1 \leq NFUS \leq 4$
3	31-33	NO	NRADX(1)	Number of points defining half section of first fuselage segment. If fuselage is circular, the program calculates the indicated number of Y and Z ordinates. $3 \leq NRADX(1) \leq 30$
3	34-36	NO	NFORX(1)	Number of stations for first fuselage segment. $4 \leq NFORX(1) \leq 20$
3	37-39	NO	NRADX(2)	Same as above for segment 2.
3	40-42	NO	NFORX(2)	Same as above for segment 2.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	43-45	NO	NRADX(3)	Same as above for segment 3.
3	46-48	NO	NFORX(3)	Same as above for segment 3.
3	49-51	NO	NRADX(4)	Same as above for segment 4.
3	52-54	NO	NFORX(4)	Same as above for segment 4.
3	55-57	NO	NP	Number of nacelles to read. NP ≤ 3.
3	58-60	NO	NPODOR	Number of stations at which nacelle radii are specified 4 ≤ NPODOR ≤ 20
3	61-63	NO	NF	Number of fins to read. NF ≤ 6.
3	64-66	NO	NFINOR	Number of ordinates defining each fin airfoil section. 3 ≤ NFINOR ≤ 10
3	67-69	NO	NCAN	Number of canards to read. NCAN ≤ 2.
3	70-72	NO	NCANOR	Number of ordinates defining each canard airfoil section. 3 ≤ NCANOR ≤ 10 If negative, airfoils are non-symmetric.
4	1-7	YES	REFA	Wing reference area
4	8-14	YES	CBAR	Pitching moment reference length. (Required for ANLZ and WDEZ only)
4	15-21	YES	XBARIN	X value of pitching moment center. (Required for ANLZ and WDEZ only)

Note: Omit this card if JO (Card 3) is 0 or 2.

Wing Description

Card Number	Card Column	Decimal Required	Variable Name	Description
5	1-70	YES	XAF	Array of percent chords at which wing airfoil ordinates will be specified. (NWAFOR values.)
6	1-7	YES	XLED	X coordinate of airfoil leading edge.
6	8-14	YES	YLED	Y coordinate of airfoil leading edge.
6	15-21	YES	ZLED	Z coordinate of airfoil leading edge.
6	22-28	YES	CLED	Airfoil chord length
	Note: This card is repeated for each airfoil, ordered inboard to outboard. (NWAF cards.)			
7	1-70	YES	TZORD	Array of camber Z values referenced to Z coordinate of airfoil leading edge, ordered leading edge to trailing edge.
	Note: This card is repeated for each airfoil, ordered inboard to outboard. Omit card set 7 if wing not cambered.			
8	1-70	YES	WAFORD	Array of airfoil upper surface half thickness ordinates expressed in percent chord, ordered leading edge to trailing edge.
	Note: Repeat Card Set 8 for each airfoil, ordered from inboard to outboard.			
	Note: Card Set 9, an option in the plot program input to define the lower surface airfoil for an asymmetric airfoil shape, was deleted from the basic geometry to reduce core size.			

Fuselage Description

Omit card sets 10-15 if J2 is 0 or 2. The fuselage is input in segments. Complete input for each segment before going on to next segment. A segment may contain ≤20 defining stations. First segment must begin at x=0.

If there is more than one fuselage segment, the first station of a segment repeats the definition of the last station of the preceding segment (i.e., cross-section is again defined at the same X station). Otherwise, a gap in the fuselage description will occur between the last station of one segment and the first station of the following segment. Make the first x value of the succeeding segment slightly larger than the last x value of the preceding segment.

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	XFUS	Array of fuselage X stations
11	1-70	YES	ZFUS	Array of Z coordinates defining fuselage centerline.
	Note:	Omit card set 11 if J6 ≠ 0 or if J2 = 1.		
12	1-70	YES	FUSARD	Array of fuselage cross sectional areas.
	Note:	Omit card set 12 if J2 not equal to -1 or 3.		
13	1-70	YES	FUSPER	Array of fuselage perimeters
	Note:	Omit card set 13 if J2 not equal to 3.		
14	1-70	YES	SFUS	Array of Y coordinates defining first station half section, ordered bottom to top.
15	1-70	YES	SFUS	Array of Z coordinates defining first station half section, ordered bottom to top.
	Note:	Repeat card sets 14 and 15 for each station in segment 1. Omit card sets 14 and 15 if J2 is not equal to 1.		
	Note:	For each fuselage segment, repeat card sets 10 thru 15.		

Nacelle Description

Card Number	Card Column	Decimal Required	Variable Name	Description
	Omit card sets 16, 17 and 18 if J3 is 0 or 2.			
16	1-7	YES	PODORX	X coordinate of origin of first nacelle
16	8-14	YES	PODORY	Y coordinate of origin of first nacelle
16	15-21	YES	PODORZ	Z coordinate of origin of first nacelle
16	22-28	YES	PODZW	Z coordinate of origin of first nacelle, referenced to local wing surface. 0., program will calculate from PODORZ +D, nacelle is located D units above local wing surface -D, nacelle is located D units below local wing surface
	Note: If PODZW ≠ 0., PODORZ is not required.			
17	1-70	YES	XPOD	Array of X coordinates, referenced to nacelle origin, at which nacelle radii will be specified.
18	1-70	YES	RPOD	Array of nacelle radii
	Note: For each nacelle, repeat card sets 16 thru 18. If PODORY is non-zero, a duplicate nacelle is located symmetrically to the X-Z plane.			

Fin Description

Card Number	Card Column	Decimal Required	Variable Name	Description
Omit card sets 19, 20 and 21 if J4 is 0 or 2.				
19	1-7	YES		X coordinate of lower fin airfoil leading edge.
19	8-14	YES		Y coordinate of lower fin airfoil leading edge.
19	15-21	YES		Z coordinate of lower fin airfoil leading edge.
19	22-28	YES		Chord length of lower airfoil
19	29-35	YES		X coordinate of upper fin airfoil leading edge.
19	36-42	YES		Y coordinate of upper fin airfoil leading edge.
19	43-49	YES		Z coordinate of upper fin airfoil leading edge.
19	50-56	YES		Chord length of upper airfoil
20	1-70	YES	XFIN	Array of percent chords, ordered leading edge to trailing edge, at which fin airfoil ordinates will be specified.
21	1-70	YES	FINORD	Array of fin airfoil half thickness ordinates expressed as percent chord.

Note: Repeat card sets 19 thru 21 for each fin.
If either Y coordinate is non-zero, a duplicate fin is located symmetrically with respect to the X-Z plane.

Canard (or Horizontal Tail) Description

Card Number	Card Column	Decimal Required	Variable Name	Description
Program identifies horizontal tail or canard by location relative to wing. Omit card sets 22-25 if J5 is 0 or 2.				
22	1-7	YES		X coordinate of inboard canard airfoil leading edge.
22	8-14	YES		Y coordinate of inboard canard airfoil leading edge.
22	15-21	YES		Z coordinate of inboard canard airfoil leading edge.
22	22-28	YES		Chord length of inboard canard airfoil
22	29-35	YES		X coordinate of outboard canard airfoil leading edge.
22	36-42	YES		Y coordinate of outboard canard airfoil leading edge.
22	43-49	YES		Z coordinate of outboard canard airfoil leading edge.
22	50-56	YES		Chord length of outboard canard airfoil
23	1-70	YES	XCAN	Array of percent chords, ordered leading edge to trailing edge, at which canard airfoil ordinates will be specified.
24	1-70	YES	CANORD	Array of canard airfoil upper surface ordinates expressed as percent chord, ordered leading edge to trailing edge.
25	1-70	YES	CANOR1	Same as above for canard airfoil lower surface.

Note: If canard is symmetric, omit card set 25.

Note: For each canard, repeat card sets 22 thru 25.

4.3 Plot Program

This program draws a picture of the configuration defined in the basic geometry, as requested by the codes on card 3.

Views of the configuration are controlled by the inputs on card 4. There will be as many drawings of the configuration as there are cards 4. Three different types of card 4 inputs are possible, for orthographic, perspective, or stacked three-view options.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			PLØT
2	1-80			Any desired title information.
3	1-7	YES	AJ1	Wing input code. 0. = Ignore wing definition. 1. = Include wing definition.
3	8-14	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.
3	15-21	YES	AJ3	Nacelle input code. 0. = Ignore nacelle definition. 1. = Include nacelle definition.
3	22-28	YES	AJ4	Fin input code. 0. = Ignore fin definition. 1. = Include fin definition.
3	29-35	YES	AJ5	Canard input code. 0. = Ignore canard definition. 1. = Include canard definition.

Card Number	Card Column	Decimal Required	Variable Name	Description
				For Orthographic Projections
4	1		HORZ	X, Y, Z for horizontal axis.
4	3		VERT	X, Y, Z for vertical axis.
4	5-7		TEST1	OUT if deletion of hidden lines required; otherwise blank.
4	8-12	YES	PHI	Roll angle in degrees.
4	13-17	YES	THETA	Pitch angle in degrees.
4	18-22	YES	PSI	Yaw angle in degrees.
4	48-52	YES	PLOTSZ	Length in inches of maximum plot dimension.
4	53-55			Punch ØRT in these columns
				For Perspective Views (See fig. 4.3-1)
4	8-12	YES	PHI	X of eye point in data coordinate system.
4	13-17	YES	THETA	Y of eye point in data coordinate system.
4	18-22	YES	PSI	Z of eye point in data coordinate system.
4	23-27	YES	XF	X of focal point in data coordinate system.
4	28-32	YES	YF	Y of focal point in data coordinate system.
4	33-37	YES	ZF	Z of focal point in data coordinate system.
4	38-42	YES	DIST	Distance from eye point to view plane in inches.
4	43-47	YES	FMAG	View plane magnification factor. Controls size of projected image.
4	48-52	YES	PLOTSZ	Diameter of view plane in inches. DIST and PLOTSZ determine a cone which is the field of vision.
4	53-55		TYPE	Punch PER in these columns

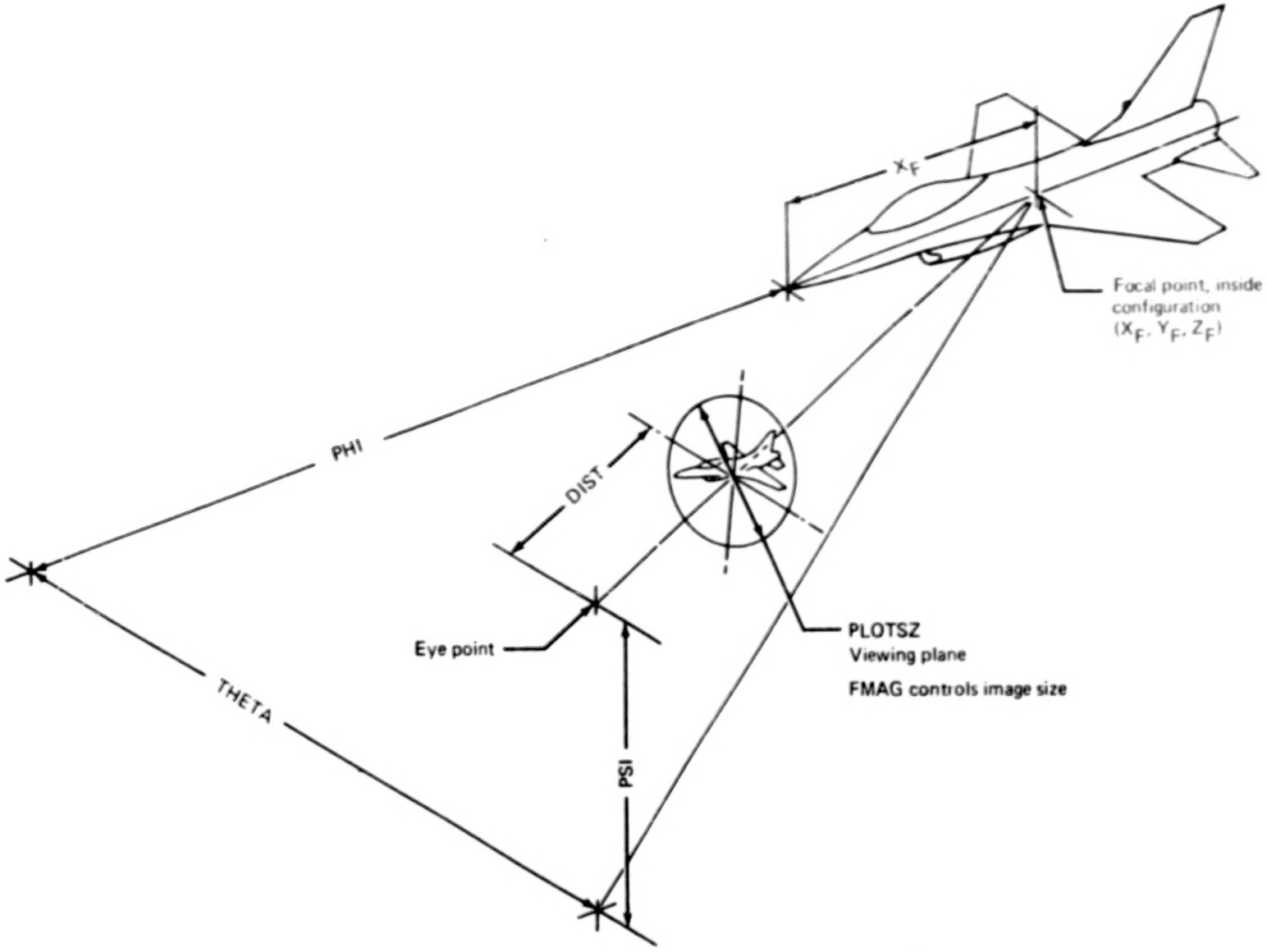

FIGURE 4.3-1.—PERSPECTIVE VIEW INPUTS

Card Number	Card Column	Decimal Required	Variable Name	Description
				For Plan, Front and Side Views (Stacked)
4	8-12	YES	PHI	Ordinate of plan view centerline on paper (in inches).
4	13-17	YES	THETA	Ordinate of side view centerline on paper (in inches).
4	18-22	YES	PSI	Ordinate of front view centerline on paper (in inches).
4	48-52	YES	PLOTSZ	Length (in inches) of maximum plot dimension.
4	53-55		TYPE	Punch VU3 in these columns
	Note:	For each additional plot desired, card 4 will be repeated at this position in the data deck.		
5	1-3			END

4.4 Skin Friction Program

Codes on card 3 control inclusion of basic geometry as requested. Where additional input is required (e.g., fuselage perimeter option), areas or lengths are input in units consistent with the basic geometry definition.

The skin friction coefficient subroutine in the program requires lengths in feet. The input lengths are converted to feet, if necessary, using the factor SCAMOD on card 5 or 6.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			SKFR
2	1-70			Any desired TITLE information

Card Number	Card Column	Decimal Required	Variable Name	Description
3	1-7	YES	AJ1	Wing input code -1. = Wing defined in basic geometry. Make no correction for wing-fuselage joint. 0. = No wing defined. 1. = Wing defined in basic geometry. Subtract wing root area from fuselage wetted area. 2. = Wing same as preceding case.
3	8-14	YES	AJ2	Fuselage input code -1. = Wetted area and reference length will be input. 0. = No fuselage defined. 1. = Fuselage defined in basic geometry. 2. = Fuselage same as preceding case.
3	15-21	YES	AJ3	Nacelle input code -1. = Wetted area and reference length will be input. 0. = No nacelles defined. 1. = Nacelles defined in basic geometry. 2. = Nacelles same as preceding case.
3	22-28	YES	AJ4	Fin input code -1. = Fins defined in basic geometry. Make no correction for fin-fuselage joint. 0. = No fins defined. 1. = Fins defined in basic geometry. Subtract fin root area from fuselage wetted area. 2. = Fins same as preceding case.
3	29-35	YES	AJ5	Canard (or horizontal tail) input code -1. = Canards defined in basic geometry. Make no correction for canard-fuselage joint. 0. = No canards defined. 1. = Canards defined in basic geometry. Subtract canard root area from fuselage wetted area. 2. = Canards same as preceding case.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	1-7	YES	AKI	Mach number-altitude combination code -AKI = Combination same as preceding case. 0 = Use Mach number-Reynolds number combinations. AKI = Number of Mach-altitude combinations. AKI ≤ 20.
4	8-14	YES	AK4	Mach number-Reynolds combination code -AK4 = Combinations same as preceding case. 0 = Use Mach number-altitude combinations. AK4 = Number of Mach-Reynolds number combinations. AK4 ≤ 20.
4	15-21	YES	AXTPT	Miscellaneous components code -AXTPT = Same components as preceding case. 0 = No miscellaneous components defined. AXTPT = Number of miscellaneous components. NXTPT ≤ 10.
4	22-28	YES	POVLP	Total overlap area for nacelles Subtract from wing wetted area.
5	1-7	YES	AM	Mach number
5	8-14	YES	AL	Altitude (feet/1000)
5	15-21	YES	DELT	Temperature deviation from standard day (^{o}F)
5	22-28	YES	SCAMOD	Scale factor to convert input dimensions to feet.

Note: There will be AKI of card(s) 5.
Omit card set 5 if AKI is 0 or negative.

Card Number	Card Column	Decimal Required	Variable Name	Description
6	1-7	YES	AM	Mach number
6	8-14	YES	RNPFL	Reynolds number per foot length x $10.^{-6}$
6	15-21	YES	SCAMOD	Scale factor to convert input dimensions to feet.
6	22-28	YES	TOTEM	Total temperature (^{o}R)
	Note:	There will be AK4 of card(s) 6. Omit card set 6 if AK4 is 0 or negative.		
7	1-7	YES	SWETRB	Fuselage wetted area
7	8-14	YES	FUSL	Fuselage reference length.
	Note:	Omit card 7 if J2 is 0, 1 or 2.		
8	1-7	YES	SWETNA	Total nacelle wetted area
8	8-14	YES	TODL	Nacelle reference length.
	Note:	Omit card 8 if J3 is 0, 1 or 2.		
9	1-7	YES	SWETXP	Wetted area of miscellaneous component.
9	8-14	YES	RXLP	Reference length of miscellaneous component.
9	15-24		PTITLE	Any desired title information.
	Note:	There will NXTPT of card(s) 9. Omit card set 9 if NXTPT is 0 or negative.		
For each new case, add Cards 2 through 9 at this position in the data deck.				
10	1-3			END

4.5 Far-Field Wave Drag Program

Codes on card 3 control inclusion of basic geometry data as requested. The case number in first field of card 4 is an integer, and must be right justified in the field, without decimal. Other input are in 10 field, -7 digit format.

If the fuselage restraint feature is used, the resulting fuselage definition for the last case will be stored and can be used to update the basic geometry (see executive control card summary, FSUP).

Multiple cases involving a given configuration description (e.g., various Mach numbers) may be run by a card 4 series. If the geometry is to be changed, an END card must be input and the program re-entered by an FFWD or GEOM and FFWD set-up.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			FFWD
2	1-80			Any desired title information.
3	1-7	YES	AJ1	Wing input code 0. = Ignore wing definition. 1. = Include wing definition.
3	8-14	YES	AJ2	Fuselage input code 0. = Ignore fuselage definition. 1. = Include fuselage definition.
3	15-21	YES	AJ3	Nacelle input code 0. = Ignore nacelle definition. 1. = Include nacelle definition.
3	22-28	YES	AJ4	Fin input code 0. = Ignore fin definition. 1. = Include fin definition.
3	29-35	YES	AJ5	Canard input code 0. = Ignore canard definition. 1. = Include canard definition.

Case Cards

Cards 4 input a series of cases of different Mach number, cut or theta variables, and/or fuselage restraints. The solution is performed with the fuselage as input, and also for an optimum fuselage shape (subject to restraint points at which the fuselage shape must be as input). If no fuselage restraint is specified (NREST = 0.), one will be assumed at the station of maximum overall area. Do not input restraint stations at nose or aft end of fuselage (those are automatically assumed). If NREST ≠ 0., a restraint card (card 5) will follow the case card. Restraint condition will apply for subsequent cases if NREST is not changed.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	1-4	NO	CASE	Case identification (right-justified)
4	8-14	YES	XMACH	Mach number
4	15-21	YES	NX	Number of equal intervals into which the portion of the X-axis, XA to XB for each roll angle, is to be divided. NX ≤ 100. and an even number
4	22-28	YES	NTHETA	Number of equal intervals into which the domain of theta (-90^0 to 90^0) is to be divided. NTHETA ≤ 36., and a multiple of 4. If the area distribution at -90^0 only is desired, input NTHETA = 1.
4	29-35	YES	NREST	Number of X stations for fuselage restraint points (≤10.), used for all subsequent cases if NREST does not change. If NREST = 0., program assumes restraint points at nose, base and station of maximum overall area.
5	1-70	YES	XREST	Array of fuselage stations (including nose and base) at which computed minimum drag curve will be restrained to input area.
	Note:	Repeat card 4 for each new case. Only 1 card 5 may be input, after first card 4 with NREST ≠ 0.		
6	1-3			END

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4.6 Near-Field Wave Drag Program

Codes on card 3 control inclusion of basic geometry as requested.

Two options are provided for fairing the wing section shape at a given spanwise station: linear or second order, controlled by TNOPCT on card 4.

The code ANYBOD (on card 5) identifies the span station of the inboard end of the wing for calculating wing thickness pressures and wave drag. This is the y value of the wing-fuselage intersection if there is a fuselage. The inboard end or intersection may be input (on card(s) 6) if ANYBOD = -10.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			NFWD
2	1-72			Any desired TITLE information.
3	1-7	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition. 1. = Include fuselage definition.
?	8-14	YES	AJ3	Nacelle input code 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
3	15-21	YES	AJ4	Fin input code. 0. = Ignore fin definitions. 1. = Include fin definitions.
3	22-28	YES	AJ5	Canard input code. 0. = Ignore canard definitions. 1. = Include canard definitions.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	1-7	YES	TNOPCT	Wing airfoil fairing code. -1. = Linear chordwise fairing. 0. = Second order fairing.
4	8-14	YES	XM	Basic Mach number for this case.
4	15-21	YES	TNOM	Number of additional Mach numbers. TNOM ≤ 5
4	22-28	YES	DONT	Wing data printout code. 0. = Minimal printout. 2. = Thickness pressure coefficients at each grid element in the wing calculations will be printed. 101. = Velocity potential will also be printed.
4	29-35	YES	TNON	Number of semispan element rows in wing calculations. TNON ≤ 40. If blank, TNON set to 40.
4	36-42	YES	TJBYMX	Number of spanwise stations at which wing thickness pressures are calculated TJBYMX ≤ 24. Leave blank if TNON not specified.
4	43-49	YES	TNCUT	Number of body stations at which pressure coefficients are calculated (≤60). If blank, TNCUT set to 50.
5	1-7	YES	ANYBOD	Wing Y dimension at inboard edge. If negative, program will solve for wing-fuselage intersection. -10.0 = Wing-body intersection will be input at x/c of basic geometry on card(s) 6.
5	8-14	YES	WRAP	Nacelle pressure field code. -1. = Wrap solution for nacelle pressure field is desired. 1. = Glance solution is performed.

Card Number	Card Column	Decimal Required	Variable Name	Description
5	15-21	YES	DLT2	Interference printout code. -1. = Summary table printout only. 1. = Details of nacelle/fuselage interference calculations will be printed.
5	22-28	YES	BCUT	Number of divisions of nacelles used to define nacelle pressures and Whitham F(Y) function. BCUT ≤ 40. If blank, BCUT set to 40.
5	29-35	YES	XXIN	Nacelle fairing option (R vs X). 0. = Second order fit. 1. = Linear fit.
				Wing-Fuselage Intersection
6A	1-70	YES	WX	X values
6B	1-70	YES	WY	Y values
6C	1-70	YES	WZ	Z values
	Note:			Input X array defining wing-fuselage intersection, then Y and Z. Start each array on a new card. Values are input at the percent chords of the basic geometry wing definition (NWAFOR values for each array). Omit cards 6 if ANYBOD ≠ -10.
7	1-35	YES	TXM	Array of additional Mach numbers. Solution will be performed for these Mach numbers after the solution for XM.
	Note:			There will be a total of TNOM values on the card. Omit this card if TNOM = 0.
8	1-70	YES	TYB2	Array of semispan values of element row at which wing thickness pressures are calculated.
	Note:			These values should be whole numbers beginning with 0. and ending with TNON. Up to ten values per card. Up to three cards. Omit these cards if TJBYMX was not specified.
9	1-3		END	

4.7 Wing Design Program

The wing design program principally requires a wing planform (supplied from the basic geometry), a description of the loadings to be used in optimizing the wing shape, and specification of the design point and constraints to be applied to the solution.

Punch all data, with decimals, to the left in the card columns (10 field, 7 digit format).

Default options are provided to help keep input simple. These include:

- TLOADS — This is the number of loadings to be used in finding an optimum loading combination. If input as a positive number, the specified number of loadings will be taken, in order, from the table on page 82. (A negative sign requires the user to list the loading numbers to be used.)
- XOCNUM — This is the number of percent chords used in printing the camber surface output. If blank, standard percent chords are used.
- TJBYMX — Standard semispan stations (TJBYS) are provided if TJBYMX = 0. and TNON = 0.

If program options are used that require wing thickness pressures, nacelle buoyancy field, fuselage upwash loading, or asymmetric fuselage loading, it is necessary to have previously run the near-field wave drag or lift analysis programs to obtain the proper tables. This is done as follows:

- Nacelle buoyancy loading — May be calculated by either wing analysis program or near-field wave drag program.
- Wing thickness pressures — Obtained from near-field wave drag program.
- Fuselage upwash loading — Obtained by running lift analysis program with wing slopes zeroed (WHUP = 1.0).
- Asymmetric fuselage loading — Obtained from lift analysis program with SYMM = 1.0 or 2.0.

The most efficient way to obtain all of the configuration-dependent data is to first run the near-field wave drag program, without nacelles, to get the wing thickness pressures. Then run the lift analysis program, with nacelles, and with the zero slope option (WHUP = 1.0) and asymmetric fuselage option (SYMM ≠ 0). The configuration-dependent data then are stored in program common blocks for use in the wing design program. Alternatively, the configuration-dependent data may be written on a file (TAPE20) in the WPLT

overlay, which can be used to create punched card decks for input into the wing design program.

The fuselage upwash loading will be that obtained with the fuselage at the specified incidence. If the upwash fields corresponding to a series of fuselage angles of attack are desired, it will be necessary to rerun the lift analysis program to produce each upwash pressure loading. Notice that the fuselage angle of attack in both the lift analysis and wing design programs can be different from the incidence in the basic geometry. In the wing design program, angle of attack is input on card 3. In the lift analysis program, angle of attack is input on card 24 as a special case of the pressure limiting option (requires FLIMIT = 1.0 on card 4 and appropriate value of VACFR on card 23).

The number of elements in the wing grid system is controlled by input TNON (normally set at 40.). Depending upon the Mach number and planform, some program dimension limits may be exceeded with TNON = 40. If this occurs, the program solves for the allowable TNON, prints it, and stops the design case. It is then necessary for the user to reduce TNON and the associated camber line variables TJBYMX and TJBYS.

There is an option to allow bypassing the optimization feature of the wing design program and obtain a camber surface corresponding to a single camber type loading (i.e., loading number less than 15 in the table on page 82). The single loading, requested by SLFLAG = 1.0 on card 4, may be used in combination with the configuration-dependent loadings (15-17) if desired, since the configuration-dependent loadings do not produce a camber surface. No constraint, other than C_L (CLDZIN on card 5) may be imposed on the solution. If CLDZIN = 0., the camber surface will be generated for the loading requested, without scaling. A particular form of the single loading feature is to input a tabular loading as the nacelle pressure field, and to then request the corresponding camber surface. If this is done, do not use the smooth form of the nacelle pressure field (code FNPSMO); also, loading 17 should not be requested since the input tabular loading would have replaced any actual nacelle pressure field.

****CAUTION****

The loading options must be used with some care. Loadings 12 and 13 cannot be used without also using loadings 15 and 16. Loading 11 cannot be used without specifying a corresponding planform region (ANOARB≠0). If all loadings are requested, the resultant optimum combination of loadings (and camber shape) may be physically unrealistic if no constraints on upper surface pressure coefficient are imposed.

If the fuselage is included in the solution, it will be necessary to also use Z constraints at the side of fuselage station to obtain a directly usable camber surface shape and a valid drag integration.

DESCRIPTION OF WING LOADING TERMS

Loading Number	Definition
1.	Uniform
2.	Proportional to x, the distance from the leading edge
3.	Proportional to y, the distance from the wing centerline
4.	Proportional to y^2
5.	Proportional to x^2
6.	Proportional to x(c - x), where c is local chord
7.	Proportional to x^2 (1.5 c -x)
8.	Proportional to $2\,(1 + 5\,\frac{x}{c})^{-0.5}$
9.	Proportional to $\eta^2\,(\eta - 1)^2$
10.	Elliptical spanwise, proportional to $\sqrt{(1 - \eta)}$
11.	Proportional to x, the distance from the leading edge of an arbitrarily defined region
12.	A camber-induced loading proportional to the body buoyancy loading
13.	A camber-induced loading proportional to the body upwash loading
14.	A camber-induced loading proportional to the nacelle buoyancy loading
15.	The body buoyancy loading
16.	The body upwash loading
17.	The nacelle buoyancy loading

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			WDEZ
2	1-70			Any desired TITLE information.
3	1-7	YES	TNON	Number of semispan elements in wing grid system. 2. ≤ TNON ≤ 50. If blank, TNON set to 40.
3	8-14	YES	TJBYMX	Number of semispan stations at which camber surface is calculated. 2. ≤TJBYMX ≤ 25. If blank, TJBYMX set to 23. Must be specified (and TJBYS input) if TNON is not 0. or 40.
3	15-21	YES	TIFAF	Flat plate calculation code. -1. = Use data from previous case. 0. = Flat plate calculation will be made. 1. = Flat plate data will be input on card 9.
3	22-28	YES	AJ1	Fuselage input code 0. = Ignore fuselage definition 1. = Include fuselage
3	29-35	YES	YSOB	Y value of wing-fuselage intersection
3	36-42	YES	ALPB	Fuselage angle of attack, deg. (relative to attitude in basic geometry).
4	1-7	YES	APRINT	Printed output code -2. = Summary output printed. -1. = Input data (except large tables) and summary output printed. 0. = Input data, output summary and camber shapes at design condition, if requested, are printed. 1. = Same as APRINT = 0., plus some diagnostic data. 2. = All input, output and diagnostic data printed.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	8-14	YES	SMOOTH	Code to determine smoothing procedure applied to camber surface longitudinal slope at each span station. 0. = No smoothing performed. 1. = Smooth-as-you-go technique used. 2. = Three-point smoothing technique used.
4	15-21	YES	RESTART	Code to determine disposition of force and moment coefficients for component and interference loadings. -1. = Data from previous case will be used. 0. = Data will be calculated by program for use in current case and subsequent cases. 1. = Data will be calculated, and also punched on cards. 2. = Data are read from card sets 20 and 21 3. = Data are read from TAPE3 (written by previous case).
				Note: See page 38 for RESTART discussion
4	22-28	YES	YSNOOT	Y value for parabolic apex tangent to wing leading edge. (Leave blank if not used.)
4	29-35	YES	FNPSMO	Code to select type of nacelle pressure field to be used in camber surface version (loading 14). 0. = Use smooth form. 1. = Use nacelle pressure field definition.
4	36-42	YES	SLFLAG	Single loading flag to bypass optimization. 0. = Loading optimization requested 1.0 = Use single camber-type loading

Card Number	Card Column	Decimal Required	Variable Name	Description
5	1-7	YES	XM	Basic Mach number.
5	8-14	YES	CMO	Design value of pitching moment coefficient at zero lift.
5	15-21	YES	CLDZIN	Value of design lift coefficient. If blank or zero, CLDZIN set to 1.0, unless SLFLAG = 1.0. (If SLFLAG = 1.0. and CLDZIN = 0., camber surface calculated for requested loading without scaling.)
5	22-28	YES	TLOADS	Number of loadings to be combined TLOADS ≤ 17. TLOADS < 0. = Loading numbers will be input on card(s) 10. Loading numbers will be taken from table on page 82. TLOADS > 0. = Loadings will be in the order tabulated on page 82. E.g., if TLOADS = 3.0, first 3 loadings from page 82 will be used.
5	29-35	YES	XOCNUM	Number of chordwise locations at which camber ordinates will be printed, corresponding to options selected on card 7. (XOCNUM) ≤ 20. 0. = Default locations of 0., 5., 10., 20., 30., ... 90., 100. are used. Omit card 11. + = Values in percent of local chord will be input (card 11).
5	36-42	YES	ANOARB	Numbers of points on card(s) 12 used to define the arbitrary region of the wing planform for loading number 11. ANOARB ≤ 20. If blank, cards 12A and 12B not read.
5	43-49	YES	AXCPLIM	Number of chordwise locations (card set 18A) used to specify wing upper surface limiting pressures. AXCPLIM ≤ 10. - = Use default set of limiting pressures. Requires card 17 to be read. 0. = Card set 18 not read. + = Card set 18 will be read.

Card Number	Card Column	Decimal Required	Variable Name	Description
5	50-56	YES	AYCPLIM	Number of spanwise stations (card set 18B) used to specify wing upper surface limiting pressures. Needed only if AXCPLIM > 0. AYCPLIM ≤ 30.
5	57-63	YES	AXCPGRD	Number of chordwise locations (card set 19A) used to specify wing upper surface pressure gradient constraints. AXCPGRD ≤ 10. Needed if AXCPLIM ≠ 0.
5	64-70	YES	AYCPGRD	Number of spanwise stations (card set 19B) used to specify wing upper surface gradient constraints. AYCPGRD ≤ 20. Needed only if AXCPGRD ≠ 0.

The following codes control input of configuration-dependent pressures. If TXCPT > 1.0, or if BODCPX, BODUPX, or CPNACX > 0., these data will be input from cards; otherwise the pressures will either not be used or will be provided by previous program execution or from the RESTART option (RESTART > 1.0). Use of the configuration-dependent pressures is controlled by TXCPT = 1.0 or by loading number selection. Card deck input not allowed if RESTART > 1.0.

Card Number	Card Column	Decimal Required	Variable Name	Description
6	1-7	YES	BODCPX	Fuselage buoyancy code. 0. = Not used or from common or RESTART. 2.0 - 11. = No. of x values input on card(s) 13A.
6	8-14	YES	BODCPY	No. of Y values corresponding to BODCPX. 0. = Not used or from common or RESTART. 2.0 - 21. = No.of y values input on card(s) 13B.
6	15-21	YES	BODUPX	Fuselage upwash code. 0. = Not used or from common or RESTART. 2.0 - 20. = No. of x values input on card(s) 14A.

Card Number	Card Column	Decimal Required	Variable Name	Description
6	22-28	YES	BODUPY	No. of values corresponding to BODUPX. 0. = Not used or from common or RESTART. 2.0 - 41. = No. of y values input on card(s) 14B.
6	29-35	YES	CPNACX	Nacelle pressure field code. 0. = Not used or from common or RESTART. 2.0 - 20. = Number of x stations for nacelle pressure field input on card(s) 15A.
6	36-42	YES	CPNACY	Number of Y values corresponding to CPNACX. 0. = Not used or from common or from RESTART. 2.0 - 30. = Number of y values input on card(s) 15B.
6	43-49	YES	TXCPT	Code to request use of wing thickness pressures in pressure limiting calculations. 0. = Wing thickness pressures not used. 1. = Wing thickness pressures used from common or from RESTART. 2.0 - 21. = Input on card(s) 16A. TXCPT = number of x values.
6	50-56	YES	TYCPT	Number of Y values corresponding to TXCPT. 0. - 1.0 = Not used or from common or from RESTART. 2.0 - 30. = Number of y values input on card(s) 16B

Solution and Constraint Options

Card 7 contains four inputs which control the extent of the solution and the constraints to be applied. Each of the 4 inputs may take on 4 different values, as follows:

0. No solution of this type desired.

1. Calculate pressure distribution, drag, and pitching moment for optimum combination of loadings.

2. Same as 1, plus also calculate the wing shape required to support the optimum pressure distribution.

3. Same as 2, plus also punch the wing shape on cards. Order is percent chords for ordinates, percent span stations, and then the ordinates in percent chord. 10F7.3 format. (May be input directly into wing analysis program with TIFZC = 1.0.)

Card Number	Card Column	Decimal Required	Variable Name	Description
7	1-7	YES	CONSTR(1)	Obtain solution for minimum drag with constraint on C_L only.
7	8-14	YES	CONSTR(2)	Obtain solution with constraints on C_L and C_{mo} (requires C_{mo} value on card 5).
7	15-21	YES	CONSTR(3)	Obtain solution with constraint on C_L and pressure limiting on wing upper surface.
7	22-28	YES	CONSTR(4)	Obtain solution with constraint on C_L and C_{mo}, plus pressure limiting on wing upper surface.
7	29-35	YES	ANZ	Number of Z constraint locations ($\leq$ 5). See discussion on page 39. -ANZ = Z locations in restart deck will be shuffled. Input cards 7D and 7E. 0. = No Z constraints ANZ = Number of Z constraints. Input cards 7A, 7B, 7C.

Card Number	Card Column	Decimal Required	Variable Name	Description
7A	1-35	YES	XZCON	X location of Z constraint.
7B	1-35	YES	YZCON	Y location of Z constraint.
7C	1-35	YES	ZCON	Z values at XZCON and YZCON.
	Note:	There will be ANZ values on cards 7A, 7B and 7C. Z values are with respect to local wing leading edge, consistent with units of basic geometry. Omit cards 7A, 7B and 7C if ANZ $\leq$ 0.		
7D	1-35	YES	NCON	Z location numbers corresponding to RESTART deck order. See discussion on page 39.
7E	1-35	YES	ZCON	Z values at location defined by NCON.
	Note:	There will be (ANZ) values on cards 7D and 7E. Z values on 7E are with respect to local wing leading edge, in same order as constraint numbers on 7D. Omit cards 7D and 7E if ANZ $\geq$ 0.		
8	1-70	YES	TJBYS	Array of semispan stations at which the camber surface is calculated.
	Note:	Up to ten values per card. There will be a total of TJBYMX whole numbers which must begin with 0.0 and end with TNON. If TJBYMX was blank, the following 23 values are used: 0., 1., 2., 4., 5., 6., 8., 10., 12., 14., 16., ..., 38., 40. Omit card(s) 8 if TJBYMX = 0.		
9	1-7	YES	XF	Wing aerodynamic center fraction.
9	8-14	YES	SCL9	Flat wing lift-curve slope (per degree), based on the reference area for force and moment coefficients.
9	15-21	YES	KF	Flat wing lift-dependent drag factor.
9	22-28	YES	AREA9	Planform area in program units.
9	29-35	YES	FACTOR	Gross wing area/reference area.
	Note:	Omit this card if TiFAF (card 3) $\leq$ 0. The data on card 9 would normally be calculated by a previous run of the same planform at the same Mach number.		

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	TLOAD	Loading numbers for use in pressure optimization. Integer numbers from 1.0 to 17.0, TLOADS (see card 5) in number, and in arbitrary order. Up to 10 values per card. Omit card(s) 10 if TLOADS > 0.
11	1-70	YES	TPCT	Array of X/C (percent of local chord) for which Z values will be interpolated at each span station. Omit card(s) 11 if XOCNUM = 0. on card 5.
12A	1-70	YES	YARB	Array of Y coordinates which define an arbitrary planform region for loading number 11.
	Note:	Up to ten values per card. Up to two cards. There will be a total of ANOARB values. If ANOARB (card 5) is blank, omit card set 12A.		
12B	1-70	YES	XARB	Array of X coordinates which define an arbitrary planform region for loading number 11.
	Note:	Up to ten values per card. Up to two cards. There will be a total of ANOARB values. If ANOARB (card 5) is blank, omit card set 12B.		

Card sets 13-16 allow optional input of configuration-dependent loadings according to codes on card 6. Do not input if RESTART > 1.0.

13A	1-70	YES	XBUOY	Array of X/C percents for fuselage buoyancy pressures. BODCPX values, ≤ 11.
13B	1-70	YES	YBUOY	Array of semispan percents for fuselage buoyancy pressures. BODCPY values, ≤ 21.
13C	1-70	YES	CPBUOY	Array of fuselage buoyancy pressures. All coefficients at a given span station are input in same order as XBUOY. Begin each semispan set on a new card and in same order as YBUOY.

Card Number	Card Column	Decimal Required	Variable Name	Description
14A	1-70	YES	XUPW	Array of x/c percents for fuselage upwash pressures. BODUPX values, ≤ 20.
14B	1-70	YES	YUPW	Array of semispan percents for fuselage upwash pressures. BODUPY values, ≤ 41.
14C	1-70	YES	CPBUPW	Array of fuselage upwash pressures. All coefficients at a given span station are input in same order as XUPW. Begin each semi-span set on new card, and in same order as YUPW.
15A	1-70	YES	XNAC	Array of x/c percents for nacelle pressure field. CPNACX values, ≤ 20. Input a set of XNAC values for each semispan station, ordered from root to tip, in same order as YNAC. Start each semi-span set on a new card.
15B	1-70	YES	YNAC	Array of semispan percents for nacelle pressure field. CPNACY values, ≤ 30.
15C	1-70	YES	CPNPF	Array of nacelle pressure field pressure coefficients, corresponding to XNAC array. Start each semi-span set on a new card in same order as YNAC.
16A	1-70	YES	XTHK	Array of x/c percents for wing thickness pressure coefficients. TXCPT values, ≤ 21.
16B	1-70	YES	YTHK	Array of semispan percents for wing thickness pressure coefficients. TYCPT values, ≤ 30.
16C	1-70	YES	CPTHK	Array of wing thickness pressure coefficients. All coefficients at a given span station are input in same order as XTHK. Begin each span station on a new card and in same order as YTHK.

Card 17. Pressure level constraint option. Input card 17 if AXCPLIM < 0. on card 5.

Card Number	Card Column	Decimal Required	Variable Name	Description
17	1-7	YES	XVAC	Fraction of vacuum CP for upper surface wing pressure coefficient (overall limit).
17	8-14	YES	SWPLE	Average inboard wing leading edge sweepback (degrees) for inboard shock pressure coefficient limit.
17	15-21	YES	DELB	Average fuselage area growth angle (degrees) for inboard shock pressure coefficient limit. + angle = area increasing (expanding body).
17	22-28	YES	XRPC	Percent of wing chord at side-of-fuselage for inboard shock limit to terminate. 0. < XRPC < 90.
17	29-35	YES	YVPC	Percent of wing semispan at which inboard shock pressure coefficient terminates. Pressure coefficient varies from inboard shock CP_{limit} to CP defined by XVAC between side of fuselage and YVPC.
18A	1-70	YES	XCPLIM	Array of chordwise locations (percent of local chord) used to define the wing upper surface limiting pressure coefficient. Needed if CONSTR(3) or (4) is ≠ 0. on card 7.

Note: There will be a total of AXCPLIM values starting with 0. and ending with 100. If AXCPLIM is not positive, omit card set 18A.

18B	1-70	YES	YCPLIM	Array of spanwise locations (percent or semispan) used to define the wing upper surface limiting pressure coefficient.

Note: Up to ten values per card.
There will be a total of AYCPLIM values starting with 0. and ending with 100. If AXCPLIM is not positive, omit card set 18B.

Card Number	Card Column	Card Required	Decimal Name	Variable	Description
18C	1-70	YES	CPLIMIT		Array of limiting pressure coefficients on the wing upper surface. All coefficients at a given semispan are input in the same order as XCPLIM. Begin each semispan set on a new card and in the same order as YCPLIM.
	Note:	Up to ten values per card. There will be a total of AXCPLIM X AYCPLIM values. If AXCPLIM is not positive, omit card set 18C.			
19A	1-70	YES	XCPGRD		Array of chordwise locations (percent of local chord) used to define the wing upper surface limiting pressure gradient. Needed if CONSTR(3) or (4) is ≠ 0 on card 7.
	Note:	There will be a total of AXCPGRD values starting with 0. and ending with 100. If AXCPLIM = 0., omit card set 19A.			
19B	1-70	YES	YCPGRD		Array of spanwise locations (percent of semispan) used to define the wing upper surface limiting pressure gradient.
	Note:	Up to ten values per card. There will be a total of AYCPGRD values starting with 0. and ending with 100. If AXCPLIM = 0., omit card set 19B.			
19C	1-70	YES	CPGRAD		Array of limiting upper surface pressure gradients (dCp/dx, where x corresponds to units input in basic configuration geometry). All coefficients at a given semispan are input in the same order as XCPGRD. Begin each semispan set on a new card and in the same order as YCPGRD.
	Note:	Up to ten values per card. There will be a total of AXCPGRD X AYCPGRD values. If AXCPLIM is 0., omit card set 19C.			

Card Number	Card Column	Card Required	Decimal Name	Variable	Description
20	1-80		TITLE		Title card of RESTART data.
21	1-80		RESTRT		Array of force and moment coefficients for component and interference loading, as punched from a previous run, for restarting program execution.
	Note:	Omit cards 20-21 if RESTART (card 4) is not equal to 2.0.			
22	1-3			END	

*The restart card sets 20-21 are printed on the output file and identified by the statement: RESTART DATA PUNCHED, DECK IMAGE FOLLOWS.

4.8 Lift Analysis Program

Codes on cards 3 and 4 control the inclusion of basic geometry data as requested. Input is in 10 field, 7 digit format.

Note that the wing camber surface may be defined in several ways, controlled by input TIFZC on card 4:

TIFZC	
0. or 1.	Input to lift analysis program on cards.
2.	Flat wing (Z = 0 everywhere).
3.	As defined by wing design program (which must have been run previously).
4.	As defined in basic geometry.

The wing camber surface input to the lift analysis program will automatically be used to update the basic geometry definition if TIFZC = 0. or 1.

By definition, a canard is required to be located forward of the wing, and a horizontal tail aft of the wing. One each is allowed, and they may both be input at the same time. Both are assumed to be mounted on the fuselage.

The fuselage definition (in basic geometry) cannot have discontinuities in z (meanline shape) between segments, when used in the lift analysis program, if the circular cross-section solution option is selected on card 3 (AJ2 = 1.0). If the arbitrary cross-section solution is requested for the fuselage (AJ2 = 2.0), card 28 is required to define location of discontinuities (if any) and set print codes.

The downwash shift options (for canard and wing downwash) are controlled by inputs on card 6. If the shift codes are left blank, the downwash will not be shifted. If the shift codes are input as LO, other shift options are available.

If the pressure limiting feature (controlled by FLIMIT on card 4) is used, it normally requires the wing thickness pressures from the nearfield wave drag program, which must have been run previously at the same Mach number. An option is provided to ignore the wing thickness pressures in the pressure limiting check, by input of VACFR as negative. (|VACFR| is used in calculation.)

All angles are input to the program in degrees.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			ANLZ
2	1-70			Any desired TITLE information.
3	1-7	YES	AJ2	Fuselage input code. 0. = Ignore fuselage definition 1. = Include fuselage definition and use circular cross-section solution 2. = Include fuselage definition and use arbitrary cross-section solution (requires input of card 28 and also digitized fuselage definition in basic geometry)
3	8-14	YES	AJ3	Nacelle input code. 0. = Ignore nacelle definitions. 1. = Include nacelle definitions.
3	15-21	YES	AJ5	Canard input code. 0. = Ignore canard definition. 1. = Include canard definition.
3	22-28	YES	AJ7	Horizontal tail input code. 0. = Ignore horizontal tail definition. 1. = Include horizontal tail definition.
4	1-7	YES	TJBYMX	Number of spanwise stations defining camber surface. TJBYMX $\leq$ 25.
4	8-14	YES	TNOPCT	Number of percent chords defining each spanwise station. TNOPCT $\leq$ 20.
4	15-21	YES	TIFZC	Code for camber surface ordinate. 0. = Z is input. 1. = Z/C (percent) is input. 2. = Flat wing option (Z = 0). 3. = Camber surface is defined in common block /CAMBER/. 4. = Use definition contained in basic geometry.

Note: If TIFZC is 2., 3., or 4., inputs TJBYMX, TNOPCT, TPCT, TYB2 and WZORD are not required.

Card Number	Card Column	Decimal Required	Variable Name	Description
4	22-28	YES	TNOM	Number of Mach numbers in addition to basic Mach number XM. TNOM ≤ 5.
4	29-35	YES	FNON	Number of semispan rows in wing grid system. FNON ≤ 40. If left blank, will be set to 40.
4	36-42	YES	FLIMIT	Limiting pressure feature code. 0. = Feature not desired. FLIMIT = Number of configuration angles of attack for solution using pressure limiting.
4	43-49	YES	XESUC	Wing leading edge suction calculation code. 0. = Do not calculate. 1.0 = Calculate L.E. suction.
4	50-56	YES	ATTLE	Attainable leading edge suction calculation code. - = Calculate. Requires input of |ATTLE| values of wing leading edge radii on cards 25. 0. = Do not calculate + = Calculate. Requires input of airfoil data on cards 26.
4	57-63	YES	RNCBAR	Reynolds number of reference chord length (CBAR), in millions. Required if ATTLE ≠ 0.
4	64-70	YES	SBNS	Slender body option code for circular fuselage cross-section. 0. = Characteristics propagated along Mach lines. 1.0 = Characteristics propagated normal to fuselage centerline.
5	1-7	YES	TNFLAP	Number of trailing edge flaps on right hand wing. TNFLAP ≤ 5.
5	8-14	YES	TNTWST	Number of values (Y in percent, and angle) to define wing twist. Relative to input wing shape. TNTWST ≤ 40.

Card Number	Card Column	Decimal Required	Variable Name	Description
5	15-21	YES	TNALP	Number of canard angles of attack ($\leq$ 5). Required if AJ5 $\neq$ 0.
5	22-28	YES	WRAP	Code for nacelle pressure field solution. -1. = Wrap 1. = Glance
5	29-35	YES	OXML	Mach number input code for nacelle pressure field calculations. 0. = Free stream Mach number used. 1. = Mach number input on card 19.
5	36-42	YES	DLT2	Nacelle pressure field calculation printout code. -1. = Summary only 1. = Detailed printout.
5	43-49	YES	BCUT	Number of cuts used to define pressure signature from nacelles. If blank, will be set to 40. BCUT $\leq$ 40.
5	50-56	YES	XNLRR	Nacelle fairing code. 0. = Second order fairing of naceile radius vs X. 1.0 = Linear fairing of R vs X.
6	1-7	YES	ANYBOD	Wing/fuselage intersection Y value. If negative, solve for intersection. If ANYBOD = -10.0, intersection will be input on card sets 14-16.
6	8-14	YES	THALP	Number of horizontal tail angles of attack. THALP $\leq$ 10. Required if AJ7 $\neq$ 0.
6	15-21	YES	SYMM	Asymmetric body volume term calculation code. 0. = Do not calculate. 1. = Calculate. 2. = Calculate using area distribution input on card sets 17 and 18.

Card Number	Card Column	Decimal Required	Variable Name	Description
6	22-28	YES	SMOGO	Smoothing code. 0. = Use 9 term smoothing. 1. = Use smoothing-as-computed pressure calculation.
6	29-35	YES	WHUP	Wing slope control code. 0. = Wing slopes calculated from input camber surface. 1. = Wing slopes = 0. (Used for fuselage upwash field.)
6	36-42	YES	FWSH	Wing downwash shift code. 0. = No shift. 1. = Shift according to DYWH.
6	43-49	YES	DYWH	Shift of wing downwash at tail. 0. = Use increment between side-of-fuselage stations of wing and tail. DYWH = Y shift increment (+ = outboard)
6	50-56	YES	FCSH	Canard downwash shift code. 0. = No shift. 1. = Shift according to DYCW and DYCH.
6	57-63	YES	DYCW	Shift of canard downwash at wing. 0. = Use increment between side-of-fuselage stations of canard and wing. DYCW = Y shift increment (+ outboard)
6	64-70	YES	DYCH	Shift of canard downwash at horizontal tail. 0. = Use increment between inboard Y stations of canard and tail. DYCH = Y shift increment (+ outboard).

Card Number	Card Column	Decimal Required	Variable Name	Description
7	1-7	YES	XM	Basic Mach number for case.
7	8-14	YES	TZSKAL	Scale factor for input Z ordinates. If blank, no scaling performed.
7	15-21	YES	CLIN(1)	Number of lift coefficients input for first Mach number (XM) at which the combined flat plate and camber pressure coefficients will be computed. (CLIN(1) ≤ 5.)
7	22-28	YES	CLIN(2)	Same as CLIN(1) for second Mach number (TMACH(1)).
7	29-35	YES	CLIN(3)	Same as CLIN(1) for third Mach number (TMACH(2)).
7	36-42	YES	CLIN(4)	Same as CLIN(1) for fourth Mach number (TMACH(3)).
7	43-49	YES	CLIN(5)	Same as CLIN(1) for fifth Mach number (TMACH(4)).
7	50-56	YES	CLIN(6)	Same as CLIN(1) for sixth Mach number (TMACH(5)).
8	1-35	YES	TMACH	Array of additional Mach numbers for this case. TNOM values. Omit this card if TNOM = 0.

Wing Camber Surface Definition

Omit card sets 9, 10 and 11 if TIFZC = 2., 3., or 4.

9	1-70	YES	TPCT	Array of chord percentages at which Z (or Z/C) ordinates are input and pressure coefficients are evaluated and output.

Note: Up to ten values per card. Up to two cards. There will be a total of TNOPCT values from 0. through 100.

Card Number	Card Column	Decimal Required	Variable Name	Description
10	1-70	YES	TYB2	Array of semispan percentages at which Z (or Z/C) ordinates are input.

Note: Up to ten values per card. There will be a total of TJBYMX values from 0. through 100.

11	1-70	YES	WZORD	Array of Z (or Z/C) ordinates of the right hand wing camber definition. All ordinates at a given semispan are input in the same order as TPCT. Begin each semispan percent on a new card and in the same order as TYB2.

Note: Up to ten values per card.
There will be a total of TPCT x TYB2 values.

Wing Twist Definition

Omit cards 12 and 13 if TNTWST = 0.

12	1-70	YES	YTWIST	Array of semispan percentages at which wing twist angles are input.
13	1-70	YES	ATWIST	Array of twist angles, in degrees, corresponding to YTWIST. A positive angle means an increase in local angle of attack. Linear interpolation is used for points between input points.

Note: Up to ten values per card. Up to four cards. TNTWST values.

Wing-Fuselage Intersection

Omit cards 14-16 if ANYBOD ≠ -10. or if there is no fuselage.

14	1-70	YES	WX	X values.
15	1-70	YES	WY	Y values.
16	1-70	YES	WZ	Z values.

Input X array defining wing-fuselage intersection, then Y and Z. Start each array on a new card. Values are input at the percent chords of the camber surface definition (card 9), or basic geometry definition (if WZORD not input).

Card Number	Card Column	Decimal Required	Variable Name	Description

Asymmetric Fuselage Area Input

Omit cards 17 and 18 if SYMM ≠ 2.0 or if there is no fuselage.

17	1-70	YES	AOVR	Above-wing area.
18	1-70	YES	AUND	Under-wing area.

Input area distribution above wing, then below. Start each array on a new card. Values are input at the percent chords of the camber surface definition (Card 9), or basic geometry definition (if WZORD not input).

Alternate Mach Nos. For
Nacelle Pressure Field Calculations

Omit card 19 if OXML = 0.

19	1-42	YES	TMLOC	Array of local Mach numbers for nacelle pressure field calculations. First value corresponds to XM, successive values correspond to TMACH (if included).

Note: Up to six values on the card.
There will be a total of TNOM + 1. values.

Wing Flap Definition

Omit cards 20 if TNFLAP = 0.

20	1-7	YES	XI	Inboard X value of flap leading edge.
20	8-14	YES	YI	Inboard Y value of flap leading edge.
20	15-21	YES	XO	Outboard X value of flap leading edge.
20	22-28	YES	YO	Outboard Y value of flap leading edge.
20	29-35	YES	DEFLAP	Flap deflection in degrees. A positive angle means the flap trailing edge is deflected downward.

Note: There will be a total of TNFLAP cards, one for each flap.

Card Number	Card Column	Decimal Required	Variable Name	Description
21	1-35	YES	TCA	Array of canard angles of attack. A positive angle means the leading edge is rotated upward.
	Note:	There will be a total of TNALP values on the card. Omit this card if TNALP = 0. or AJ5 = 0.		
22	1-63	YES	THA	Array of horizontal tail angles of attack. A positive angle means the leading edge is rotated upward.
	Note:	There will be a total of THALP values on the card. Omit this card if THALP = 0. or AJ7 = 0.		
23	1-7	YES	VACFR	Fraction of vacuum pressure coefficient for pressure limiting. If VACFR is input as negative, thickness pressures will not be included in pressure limiting calculation.
24	1-35	YES	TLALP	Array of α's for limiting pressure coefficient.
	Note:	There will be a total of FLIMIT values on the card. Omit cards 23 and 24 if FLIMIT = 0.		
		Airfoil Data for Attainable Leading-Edge Suction		
25A	1-70	YES	TYR	Array of semispan percents for airfoil leading edge radii definition. |ATTLE| values, $\pm$20, from 0. through 100.
25B	1-70	YES	TRLE	Array of airfoil leading-edge radii, R/chord, in percent, corresponding to TYR.
		Input cards 25 if ATTLE is negative. Up to 10 values per card. Start each array on new card.		
26A	1-70	YES	TY	Array of semispan percents for airfoil definitions. ATTLE values, $\pm$20., from 0. through 100.
26B	1-70	YES	TETA	Array of chord percentages for locations of maximum airfoil thickness. Input in same order as TY.

Card Number	Card Column	Decimal Required	Variable Name	Description
26C	1-70	YES	TTOC	Array of airfoil maximum thickness ratio, t/c, in percent, corresponding to TY. (This is total airfoil t/c, not half thickness as input in basic geometry.)
26D	1-70	YES	TRLE	Array of airfoil leading-edge radii, R/chord, in percent, corresponding to TY.

Input cards 26 if ATTLE is positive. Up to 10 values per card. ATTLE values per array. Start each array on a new card.

27	1-35	YES	CLINP	Arrays of lift coefficients for the input Mach numbers (XM and TMACH) at which the combined flat plate and camber pressure coefficients are computed. C_L's for each Mach number begin on a new card.

Note: Up to five values per card. Up to six cards. The number of values on each card will correspond with CLIN(I) on card 7.
If CLIN(I) = 0., omit the Ith card.

Arbitrary Fuselage Solution Data (card 28).
Input card 28 if AJ2 = 2.0 on card 3.

28	1-7	YES	D12	Print code for fuselage surface pressures. 0. = No print. 1.0 = Print.
28	8-14	YES	D13	Print field point pressures and perturbation velocities. 0. = No print. 1.0 = Print.
28	15-21	YES	D14	Print axisymmetric solution geometry, source characteristics and reference axis data. 0. = No print. 1.0 = Print.

Card Number	Card Column	Decimal Required	Variable Name	Description
28	22-28	YES	D15	Print interpolated contour data for fuselage at field point stations. 0. = No print. 1.0 = Print.
28	29-35	YES	DISC(2)	Fuselage segment discontinuity code for segment no. 2. 0. = No discontinuity at beginning of segment 2. 1.0 = Discontinuity at beginning of segment 2.
28	36-42	YES	DISC(3)	Fuselage segment discontinuity code for segment no. 3. 0. = No discontinuity at beginning of segment 3.. 1.0 = Discontinuity at beginning of segment 3.
28	43-49	YES	DISC(4)	Fuselage segment discontinuity code for segment no. 4. 0. = No discontinuity at beginning of segment 4. 1.0 = Discontinuity at beginning of segment 4.
		For a new case, input cards 2 through 28 at this place in the data deck.		
29	1-3			END

4.9 Wing Pressure Summary Program

This program summarizes and tabulates wing upper and lower surface pressure data. Inclusion of different components of surface pressure (thickness, lift, nacelles) are controlled by codes on card 3. The program can also be used to write data on TAPE20 for the different components for input into the WDEZ program. Wing pressures are printed on output and also saved on TAPE19. See pages 54, 55 for description of TAPE19 and TAPE20 formats. Input is in 10 field, 7 digit format.

Card Number	Card Column	Decimal Required	Variable Name	Description
1	1-4			WPLT
2	1-70			Any desired TITLE information.
3	1-7	YES	DWL	Wing lifting pressures code. 0. = Do not include. 1.0 = Include. 2.0 = Also write wing lifting pressures at basic alpha onto TAPE20.
3	8-14	YES	DWT	Wing thickness pressures code. 0. = Do not include. 1.0 = Include. 2.0 = Also write wing thickness pressures onto TAPE20.
3	15-21	YES	DN	Nacelle pressure field code. 0. = Do not include. 1.0 = Include. 2.0 = Also write nacelle pressure field below wing onto TAPE20.
3	22-28	YES	DF	Fuselage pressure field code. 0. = Do not include. 1.0 = Include. 2.0 = Also write fuselage pressure field onto TAPE20.

Card Number	Card Column	Decimal Required	Variable Name	Description
3	29-35	YES	DPRINT	Basic pressure data output code. 0. = Do not print or write on TAPE20. 1.0 = Print component pressure fields and also write on TAPE20 (according to codes DWL, DWT, DN and DF).
3	36-42	YES	DLINE	Number of lines of pressure summary data per page on output. (Default = 53.)
4	1-7	YES	COND	Number of C_L or α conditions for pressure summaries. -COND = No. of alpha values ($\leq$ 10.). 0. = No lifting condition (component pressure summary or write TAPE20 only). +COND = No. of C_L values ($\leq$ 10.).
4	8-14	YES	AXPC	Number of percent chords for wing pressure summary ($\leq$ 40.) 0. = Use component pressure percent chords defined by AWPC. 2.-40. = Input AXPC values on card(s) 6.
4	15-21	YES	AYPC	Number of percent semispans for wing pressure summary ($\leq$ 50.). 0. = Use component pressure percent semispans defined by AWPC. 1.-50. = Input AYPC values on card(s) 7.
4	22-28	YES	AWPC	Code to select percent chord and/or percent semispan stations of component loading for pressure output tabulation (AXPC and/or AYPC = 0.). 0. = Not used. 1. = Use wing lifting pressure stations. 2. = Use wing thickness pressure stations. 3. = Use nacelle pressure field stations. 4. = Use fuselage pressure field stations.

Card Number	Card Column	Decimal Required	Variable Name	Description
5	1-70	YES	TBCOND	Array of C_L or α values corresponding to COND on card 4. Do not input if COND = 0.
6	1-70	YES	ATXPC	Array of percent chord values for pressure tabulation ($\leq$ 40.). Do not input if AXPC = 0.
7	1-70	YES	ATYPC	Array of percent span values for pressure tabulation ($\leq$ 50.). Do not input if AYPC = 0.
		For a new case, input cards 2 through 7 at this place in the data deck. Otherwise, input card 8 to end WPLT.		
8	1-3			END

APPENDIX

INTERACTIVE GRAPHICS

The cathode ray tube (CRT) display and program coding for the design and analysis system are based on the NASA-LRC CRT and associated software. However, all display portions of the system coding are subroutined or overlaid from the basic programs, so that the system can be readily converted to other CRT arrangements.

The basic input parameter required to activate the graphics routines is the executive card CRT (punched in columns 1-3), which may be placed at the beginning of the data deck, or anywhere else in the input that an executive card may be read. If the CRT card does not appear in the data deck, no graphic displays will be generated.

The CRT card is actually an on-off device. Successive readings of the CRT card either turn on the graphics, or turn them off and place an end-of-file mark on the hard-copy file, depending upon the previous status of the graphics routines. However, the usual mode of graphics operation is to place a CRT card at the beginning of the data deck, if graphics are desired.

BASIC CRT OPERATION

Several types of video displays are generated by the design and analysis system, using the NASA-LRC software. These include:

- Menus
  A list of display choices with corresponding function keys
- Edit tables
  A list of numbers with variable names
- Plots
  Displays of x-y plots

When a display is complete, one of two system messages will appear at the top of the video screen. If the display is a menu, the message AWAITING OPERATOR ACTION will appear. To continue processing, the user must press a defined function key, selected from the menu. The second system message is PLOT FRAME COMPLETE. When this message appears, the graphics software is programmed to allow (a) editing of the display variables, (b) resumption of program execution, or (c) hard copy plot generation. If (b) is selected, the user presses function key 3 (NEXT FRAME). Editing and hard copy options are discussed on pages 110 and 111.

Menus

Menus consist of a set of display choices, together with defined function keys. Some menu lines display sets of function keys. For instance, a menu line may say FN KEY 6 DISPLAYS WING THICKNESS. Pressing key 6 will bring up a second menu, with the message FN KEYS 1 THRU 20 IDENTIFY AIRFOIL NUMBER, which would require the user to select one of the input airfoils. It should be noted that the upper key number (20) is the maximum number of airfoils allowed in the input. For a particular case, however, the user may have input only 7 airfoils. If the user now presses an undefined function key (8 thru 54) the message AWAITING OPERATOR ACTION will appear and another function button must be chosen.

Edit Tables

If the display contains an edit table, the user may now use the console keyboard to type in a new value for any variable in the table. The variable name used on the display is first typed, followed by an equal sign and the new value, followed by the console keys RETURN and EOM. (e.g., CONSTR(3) = -1.0 will change the third value of the array CONSTR to -1.0). The new value will be displayed at the top left of the video screen.

The LRC software allows the definition of only one edit format per display in the using program. It can happen that there are both fixed and floating point numbers on the screen to be edited. If this happens, the edit format can be changed by the typed-in message FORMAT = XXX RETURN EOM (where XXX is the desired format). This format remains in effect so long as the display is up, i.e., until key 3 is pressed. In case of doubt, the display will identify the current format if the message FORMAT = RETURN EOM is typed.

Special Usage of Key 55

Function key 55 is used in two ways. If the statement "RESUMES EXECUTION" appears on the menu line and key 55 is selected, the current graphic program will be terminated and execution will continue at the next executable statement encountered. If the statement "DISPLAYS PROGRAM OPTION MENU" appears on the menu line, and key 55 is selected, the current menu will be erased and the previous menu redisplayed.

Hard-Copy Plots

Each time the system message "PLOT FRAME COMPLETE" appears on the display screen, the user has the option of generating Varian hard-copy plots of the current display, assuming the run terminates normally and the job control cards specify the correct post-processor. Selecting key 6 (RECORD PLOT) or key 3 (RECORD PICTURE) followed by key 3 (NEXT FRAME) will save the display information and continue processing.

GRAPHICS USAGE

The principal uses of the graphics routines in the design and analysis system are to display the configuration, edit input geometry and to display and/or alter the basic program calculations.

There is no provision in the system to alter the input data stream on-line, so the intended usage of the graphics and the input data card set up must be carefully coordinated. Limited capability to redirect the system calculation sequence is available and these options are displayed on the CRT screen when encountered.

Geometry

Configuration geometry may be displayed either from the PLOT module, or, in simplified form, from the geometry module, The PLOT display draws a picture of the configuration on the screen (as instructed by the input view cards), but has no edit capability. All editing of geometry must be performed in the geometry module.

When the geometry module is entered from the executive to read or change configuration geometry (executive cards GEOM, GEOM NEW, FSUP or WGUP), the CRT program DISGEOM is used to display and/or edit the configuration geometry. The first menu generated gives the user the option of executing or bypassing the video displays:

```
FN KEY    1 DISPLAY AND EDITS GEOMETRY
FN KEY   55 RESUME EXECUTION
```

When key 1 is selected, the program option menu appears:

```
FN KEY    1 DISPLAYS CONFIGURATION PLANFORM
FN KEY    2 DISPLAYS FUSELAGE AREA VS X
FN KEY    3 DISPLAYS WING CAMBER (Z vs X)
FN KEY    4 DISPLAYS WING CAMBER (Z vs Y)
FN KEY    5 DISPLAYS WING CAMBER (Z/C vs Y)
FN KEY    6 DISPLAYS WING THICKNESS (Z/C vs X/C)
```

```
FN KEY    7 DISPLAYS FUSELAGE SECTIONS (NON-CIRCULAR)
FN KEY    8 EDITS CONFIGURATION CODES
FN KEY    9 EDITS PERCENT CHORD ARRAY
FN KEY   10 EDITS X,Y,Z AND CHORD (AIRFOILS 1-10)
FN KEY   11 EDITS X,Y,Z AND CHORD (AIRFOILS 11-20)
FN KEY   12 EDIT/DISPLAY WING T.E. (TZORD)
FN KEY   13 EDIT/DISPLAY WING T.E. (TZORD + ZLE)
FN KEY   14 EDIT/DISPLAY WING THICKNESS (Z/C vs X/C)
FN KEY   15 EDITS FUSELAGE X ARRAY
FN KEY   16 EDITS FUSELAGE Z ARRAY
FN KEY   17 EDITS FUSELAGE AREA ARRAY
FN KEY   18 EDITS X,Y,Z AND D OF NACELLES
FN KEY   19 EDITS NACELLE X ARRAY
FN KEY   20 EDITS NACELLE R ARRAY
FN KEY   21 EDITS X,Y,Z AND CHORD OF FIN AIRFOILS
FN KEY   22 EDITS X,Y,Z AND CHORD OF CANARD AIRFOILS
FN KEY   23 EDITS CAMBER Z ARRAY
FN KEY   55 RESUMES EXECUTION
```

The table below describes the function of each key.

KEY	FUNCTION
1	A plan view of the configuration geometry is displayed.
2	A plot of fuselage area versus station is displayed
3	Given an airfoil number 1 through 20 (1 being most inboard) a side view plot of camber (camber value + Z of leading edge) versus station at the y of the specified airfoil is displayed.
4	Given a percent chord number 1 through 21 (1 at leading edge), a rear view plot of camber (camber value + Z of leading edge) versus y at the percent chord specified, is displayed.
5	Same as key 4 but camber value versus Y
6	Given an airfoil number 1 through 20 (1 being most inboard), a side view plot of airfoil half thickness (upper and lower) versus percent chord at the specified airfoil, is displayed. The array of thicknesses (THK) is displayed below the plot and may be edited by the user. THK (1) represents the half thickness at the leading edge.
7	Given the fuselage segment number 1 through 4, and the section number 1 through 30 within the segment, the Y and Z coordinates defining the fuselage half-section are displayed. The horizontal X axis is positioned vertically at the fuselage centerline Z value (ZFUS).

8	The basic geometry input parameters JO through XBARIN are displayed on the screen and may be edited by the user. The program defined format is I4. If it is necessary to modify variables REFA, CBAR or XBARIN, the user must first change the format to floating point, such as F8.4.
9	The percent chord array (XAF) is displayed on the screen and may be edited by the user.
10/11	Four arrays, XLED, YLED, ZLED and CLED representing the X, Y and Z coordinates of the input airfoil locations of the wing leading edge and the airfoil chord lengths are displayed on the screen and may be edited by the user. Key 10 displays coordinates of first 10 airfoils and key 11 the last 10 airfoils.
12/13	Keys 12 and 13 provide a special capability to remove "spikes" or irregularities in the wing camber surface. A plot of camberline Z values (from array WZORD) or Z + Z_{LE} versus Y along the wing trailing edge is displayed. The corresponding table of Z or Z +Z_{LE} values is displayed in a table under the plot, which may be edited. When the NEXT FRAME key is depressed, the following menu appears:

FN KEYS 1 THRU 21 DISPLAY PERCENT CHORD LINES
TRAILING EDGE MAY BE EDITED
FN KEY 33 TWISTS WING TO MATCH EDITED T. E.
FN KEY 34 RESTARTS WITH ORIGINAL CAMBER DEFINITION
FN KEY 44 SAVES NEW CAMBER DEFINITION
FN KEY 55 DISPLAYS PROGRAM OPTION MENU

KEY	FUNCTION
1-21	A plot of Z or Z + Z_{LE} versus Y at the percent chord selected is displayed.
21	The Z or Z + Z_{LE} array is displayed below the plot and may be edited.
33	If the trailing edge has been edited, the remainder of the camber surface definition is altered, by linear twist, to agree with the trailing edge change. The trailing edge is redisplayed.
34	Restart option. If the change to the trailing edge was made incorrectly, the original camber surface may be recalled and the editing redone. (The restart option is available until key 44 is depressed.)
44	The wing camber surface, WZORD, which was altered in a scratch array until now, is permanently changed to match the surface displayed under key 33.
55	Return to redisplay complete option menu.

14 Given an airfoil number 1 through 20, a side view plot of airfoil thickness versus percent chord is displayed. The thickness array of the specified airfoil is also displayed below the plot and may be edited.

17 Given a fuselage segment number 1 through 4, the array (A) of fuselage area values for the segment are displayed and may be edited.

18 Four arrays, X, Y, Z and D, representing the coordinates of the nacelle origins are displayed and may be edited.

19 Given a nacelle number 1 through 9, the array of nacelle X coordinates are displayed and may be edited.

20 Given a nacelle number 1 through 9, the array (R) of nacelle radii values are displayed and may be edited.

21 Given a fin number 1 through 6, the variables XL, YL, ZL, CL, XU, YU, ZU and CU, representing the X,Y,Z and chord lengths of the lower and upper fin airfoils are displayed and may be edited.

22 Given a canard number 1 or 2, the variables XI, YI, ZI, CI, XO, YO, ZO and CO, representing the X,Y, Z and chord lengths of the inboard and outboard canard airfoils are displayed and may be edited.

23 Given an airfoil number 1 through 20, the array (C) of camber values for the airfoil are displayed and may be edited.

Skin Friction Module/Near-Field Wave Drag Module

When the skin friction program executes, the force coefficient summary from the program may be seen, or bypassed, according to the menu below:

FN KEY 1 DISPLAYS SKIN FRICTION RESULTS
FN KEY 55 RESUMES EXECUTION

Similarily, function keys 1 and 55 display or bypass the summary results from the near-field program when it executes.

When the far-field wave drag program executes, the menu choice of display or bypass first comes up. If display (FN key 1) is selected, the display program (DIS080) will give the user the option of generating displays as follows:

FN KEY	1	ERASES SCREEN
FN KEY	2	DISPLAYS GRID
FN KEY	3	DISPLAYS BODY AREA VS X
FN KEY	4	DISPLAYS OPTIMUM BODY AREA VS X
FN KEY	5	DISPLAYS CONFIG AREA VS X
FN KEY	6	DISPLAYS RESTRAINED CONFIG AREA VS X
FN KEY	7	INTERRUPT PROGRAM TO ALLOW HARD COPY PLOT GENERATION
FN KEY	9	DISPLAYS FAR FIELD WAVE DRAG SUMMATION
FN KEY	55	RESUMES EXECUTION

The user's options at this point are different from the other displays. Here the user constructs the plot to include as many curves as desired, with or without a grid, and may or may not generate a hard copy plot. To view the configuration area plot, the user need only select key 5 (followed by key 1 to remove the plot). If the user wants a hard copy plot of all curves with a grid, he selects keys 2,3,4,5,6 and 7 followed by keys 6 or 8 and 3 (NEXT FRAME). He may then resume execution, display the drag summation or build a new display after erasing the current display with key 1.

If the user selects key 8, the menu is erased and the wave drag program drag summary is printed (illustrated by typical values):

70 CHARACTER TITLE ARRAY FOR CURRENT CASE

CASE= 14 MACH= 2.700 NX= 50 NTHETA= 36

WING VOLUME CHECK

EXACT VØLUME =	11432.023
EQUIVALENT BØDY VOLUME =	11429.954

ENTIRE AIRCRAFT

D/Q =	20.27199
CDW =	.00263
ØPT. CDW* =	.002481

DRAG OF TRANSFERRED AREA DISTRIBUTIONS

ØPTIMUM EQ. BØDY CDW* =	.00089225
AVERAGE EQ. BØDY CDW* =	.00104445
POTENTIAL CDW* CHANGE =	-.00015220

At this point, the system message PLOT FRAME COMPLETE will appear. To get a hard copy plot of the display, press key 6 or 8.

To continue, the user selects key 3, which erases the screen and re-displays the function key menu.

NOTE: There is one instance when the wave drag display subroutine will not be called. That is when the restraint points exceed allowable storage of 33, which causes the optimization calculations to be omitted.

Wing Design Module

The graphics capability of the wing design program consists of:

- Display of "bucket" plot, drag-due-to-lift factor (K_E) versus Cmo.
- K_E versus Cmo for camber surface constraint solutions, if requested
- Editing of the design solution variables (C_{mo}, C_L) and constraint or restart codes
- Continuation to next input case or return to executive

The design camber surface, which is automatically stored in common block CAMBER, can be viewed in the geometry module, but not in the wing design module.

The initial display to appear in the wing design module is the bypass or display menu:

FN KEY 1 DISPLAYS BUCKET PLOT
FN KEY 55 RESUMES EXECUTION

When key 1 is selected, the optimum drag-due-to-lift versus C_{mo} "bucket" plot is displayed. Additional symbols are also plotted, giving the flat wing (+), uniform load (x), and three term (Δ) solutions. (The uniform load and three term solutions will be plotted only if those solutions have been calculated.)

Symbols (θ) are then plotted, corresponding to solutions from the constraint options 1, 2, 3, and 4, if requested. And finally, up to 10 symbols (θ) are plotted giving the option 4 solutions from previous design cases (if the current case is one of a series of wing design cases).

After the bucket plot is generated, the NEXT FRAME key beings up the set of current design inputs:

70 CHARACTER TITLE OF CURRENT CASE

```
          CMO =  .0200
       CLDEIN =  .1000
      RESTART = 2.0000
  CONSTR(1) = 1.0000
  CONSTR(2) = 1.0000
  CONSTR(3) = 1.0000
  CONSTR(4) = 1.0000
```

The user may edit any of the variables on the display. If editing is performed, the wing design case may then be re-executed when the NEXT FRAME key is again depressed, which generates the menu:

```
FN KEY     1  EXECUTES NEXT CASE
FN KEY    55  CALCULATES EDITED DESIGN POINT
```

If key 55 is selected, the program returns to the wing optimization overlay, and recalculates the wing design for the edited design inputs. If key 1 is selected, the program continues to the next statement in the normal execution process.

When the wing design case is completed, and key 1 is selected, a final option menu is displayed:

```
FN KEY  1     TERMINATES WING DESIGN PROGRAM
FN KEY 55     READS NEXT DATA CASE
```

The purpose of this choice is to permit the user to abort a series of wing design input cases once the desired wing design has been obtained.

Lift Analysis Module

Graphics options provided in the analysis module consist of:

- Display and editing of wing twist array
- Editing of configuration angle of attack, and canard and horizontal tail setting (if used)
- Editing of Mach number, and inputs SYMM, WHUP, and ANYBOD
- Display of wing pressure coefficients and fuselage upwash
- Display of force coefficient summary

The initial menu seen is:

```
FN KEY     1  DISPLAYS WING TWIST (DEG) VERSUS SPAN
FN KEY     2  EDITS WING TWIST ARRAY
FN KEY     3  EDITS CANARD ANGLES OF ATTACK
FN KEY     4  EDITS SYMM, WHUP and ANYBOD
FN KEY    55  RESUMES EXECUTION
```

The user selects the function key associated with the task desired, noting the following conditions:

1. If function key 1 is selected and no twist array was input, no plot will be generated, and the user will be required to select another function key.

2. If function key 2 is selected, the variable TWISTN (the current number of twist angles in the array ATWIST) and the ATWIST array are displayed. If entries are added or deleted in ATWIST, a corresponding change must be made in TWISTN.

3. If function key 3 is selected, the variable ALPN (the current number of canard angles of attack in array TCA) and the TCA array are displayed. If entries are added or deleted in TCA, a corresponding change must be made to ALPN.

When key 55 is selected, the analysis module continues execution, halting with the menu,

```
FN KEY    1   DISPLAYS UPWASH VERSUS PERCENT CHORD
FN KEY    2   DISPLAYS UPWASH VERSUS PERCENT SEMISPAN
FN KEY    3   DISPLAYS WING PRESSURE VERSUS PERCENT CHORD
FN KEY    4   DISPLAYS WING PRESSURE VERSUS PERCENT SEMISPAN
FN KEY    55  RESUMES EXECUTION
```

which provides the display options indicated.

Selection of keys 1 through 4 brings up one of the following secondary menus:

```
FN KEY     1  THRU 21 IDENTIFY SEMISPAN PERCENT
FN KEY    55  DISPLAYS PROGRAM OPTION MENU

FN KEYS    1  THRU 11 IDENTIFY PERCENT CHORD
FN KEY    55  DISPLAYS PROGRAM OPTION MENU

FN KEYS    1  THRU 41 IDENTIFY SEMISPAN PERCENT
FN KEY    55  DISPLAYS PROGRAM OPTION MENU

FN KEYS    1  THRU 20 IDENTIFY PERCENT CHORD
FN KEY    55  DISPLAYS PROGRAM OPTION MENU
```

If no fuselage was input, function keys 1 or 2 will produce no response. Key 55 returns to the primary menu.

Upon resumption of the analysis calculations, program FINISH is entered which halts with the menu:

```
FN KEY     1  DISPLAY DRAG DUE TO LIFT PROGRAM RESULTS AND EDIT NEXT
              HORIZONTAL TAIL ANGLE
FN KEY    55  RESUMES EXECUTION
```

If key 1 is selected, the drag summary table is printed (illustrated with typical values):

```
                    70 CHARACTER TITLE FOR CURRENT CASE

                    MACH NUMBER =                2.70
                    CONFIGURATION ALPHA =        0.00
                    CANARD ALPHA =               0.00
                    HORIZONTAL TAIL ALPHA =      0.00

   CL    CD(OFF)        CM(OFF)        CD(ON)      CM(ON)
  .00    .000551         .00801       .000645     .00598
  .01    .000451         .00677       .000525     .00474
  .02    .000480         .00554       .000533     .00351
   .       .               .             .           .
   .       .               .             .           .
   .       .               .             .           .
  .18    .018392        -.01424       .018121     .01627
  .19    .020603        -.01548       .020311     .01751
  .20    .022942        -.01671       .022639     .01874

  NEXT HORIZONTAL TAIL ALPHA (THALP) = 1.50
```

In the table the (off) and (on) refer to nacelles. Canard alpha and horizontal tail alphas are not printed if no cannard or horizontal tail is present.

It is possible to trim the configuration by the proper selection of horizontal tail angle. If there will be another horizontal tail angle, its value is indicated as shown. The value may be edited by typing THALP = XXX RETURN EOM. Key 3 (NEXT FRAME) will then resume execution.

A broader editing capability for altering the calculation sequence is enabled by the next menus to appear. The primary menu sets up the choice:

FN KEY	1	ALLOWS USER TO VIEW AND EDIT MACH NUMBER, CONFIGURATION ALPHA AND CANARD ALPHA, FOR CURRENT AND NEXT EXECUTION CYCLE
FN KEY	55	RESUMES EXECUTION

Selection of key 1 displays the current Mach number, configuration alpha and canard alpha. In addition, it displays the next parameter in the cycle to change, which may be edited by the user (typical values are shown):

CURRENT MACH NUMBER	=	2.14
CURRENT CONFIGURATION ALPHA	=	1.70
CURRENT CANARD ALPHA	=	.50
NEXT CANARD ALPHA (CAN)	=	.95
- or -		
NEXT CONFIGURATION ALPHA (CON)	=	1.87
- or -		
NEXT MACH NUMBER (XMCH)	=	2.30

The program execution sequence is canard alpha loop, configuration alpha loop and Mach number loop, in that order. When the individual loops are complete, the words CURRENT and NEXT are replaced with LAST.

The user has the option of editing the variables CAN, CON and XMCH when they appear on the screen.

REFERENCES

1. Sommer, Simon C.; and Short, Barbara J.: Free-Flight Measurements of Turbulent-Boundary-Layer Skin Friction in the Presence of Severe Aerodynamic Heating at Mach Numbers From 2.8 to 7.0. NACA TN 3391, 1955.

2. Craidon, Charlotte B.: Description of a Digital Computer Program for Airplane Configuration Plots. NASA TM X-2074, 1970.

3. Harris, Roy V., Jr.: An Analysis and Correlation of Aircraft Wave Drag. NASA TM X-947, 1964.

4. Carlson, Harry W.; and Middleton, Wilbur D.: A Numerical Method for the Design of Camber Surfaces of Supersonic Wings With Arbitrary Planforms. NASA TN D-2341, 1964.

5. Sorrells, Russell B.; and Miller, David S.: Numerical Method for Design of Minimum-Drag Supersonic Wing Camber With Constraints on Pitching Moment and Surface Deformation. NASA TN D-7097, 1972.

6. Kulfan, R. M.; and Sigalla, A.: Real Flow Limitations in Supersonic Airplane Design. AIAA Paper 78-147R, 1978.

7. Carlson, Harry W.; Mack, Robert J.; and Barger, Raymond L.: Estimation of Attainable Leading-Edge Thrust for Wings at Subsonic and Supersonic Speeds. NASA TP-1500, 1979.

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15. Supplementary Notes

Langley Technical Monitor: B. L. Shrout
Part 2 of final report

16. Abstract

An integrated system of computer programs has been developed for the design and analysis of supersonic configurations. The system uses linearized theory methods for the calculation of surface pressures and supersonic area rule concepts in combination with linearized theory for calculation of aerodynamic force coefficients. Interactive graphics are optional at the user's request.

The description of the design and analysis system is broken into four parts, covered in four separate documents:

Part 1 - General Description and Theoretical Development (NASA CR-3351)
Part 2 - User's Manual (NASA CR-3352)
Part 3 - Computer Program Description (NASA CR-3353)
Part 4 - Test Cases (NASA CR-3354).

This part contains a user's guide to the program and describes the program input.

These four documents supersede NASA contractor reports CR-2715, CR-2716, and CR-2717, which described an earlier version of the system.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
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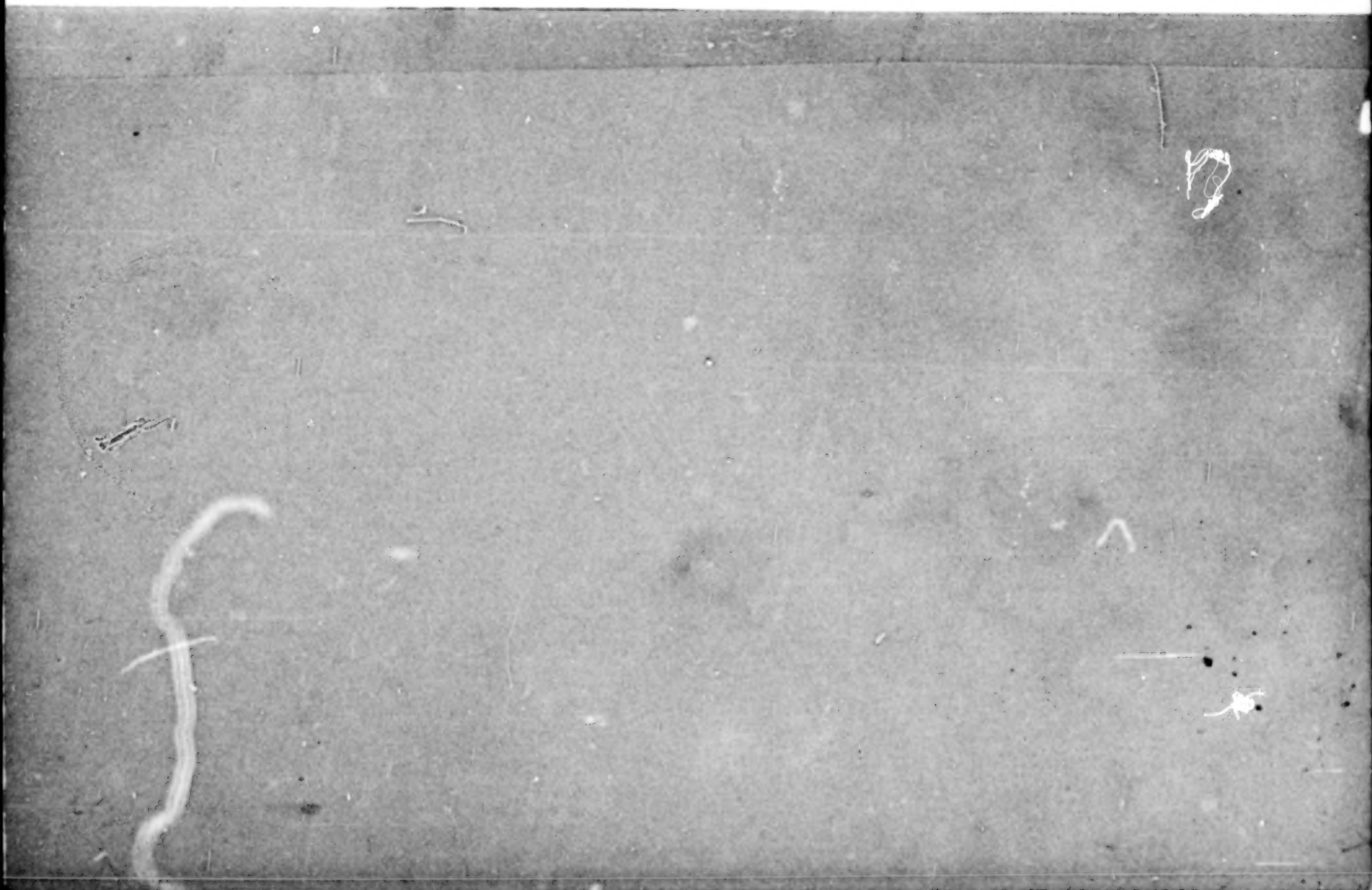

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March 2, 1981